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LANDSAT D INSTRUMENT MODULE STUDY

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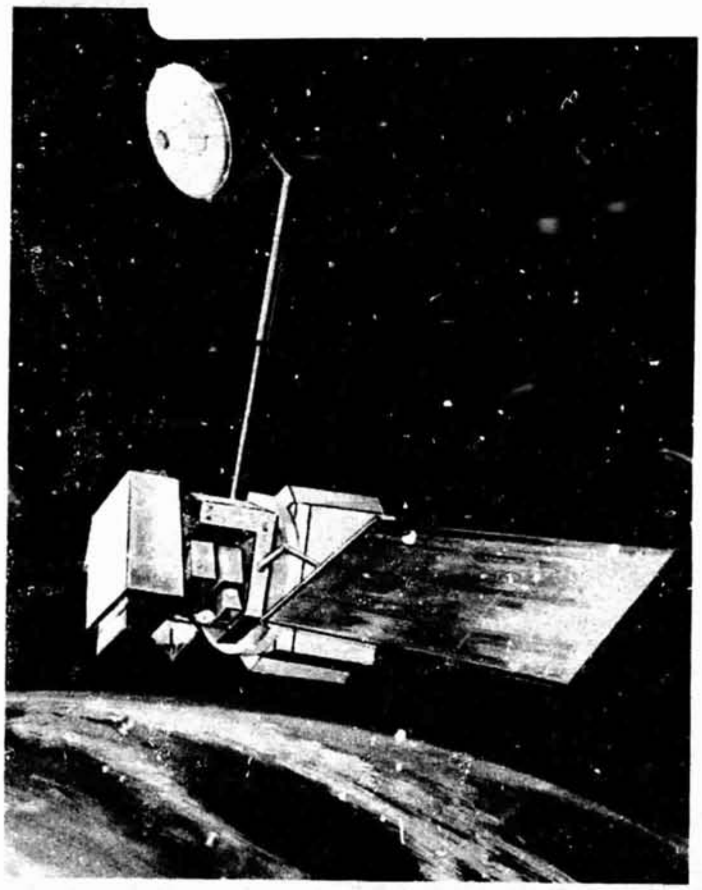

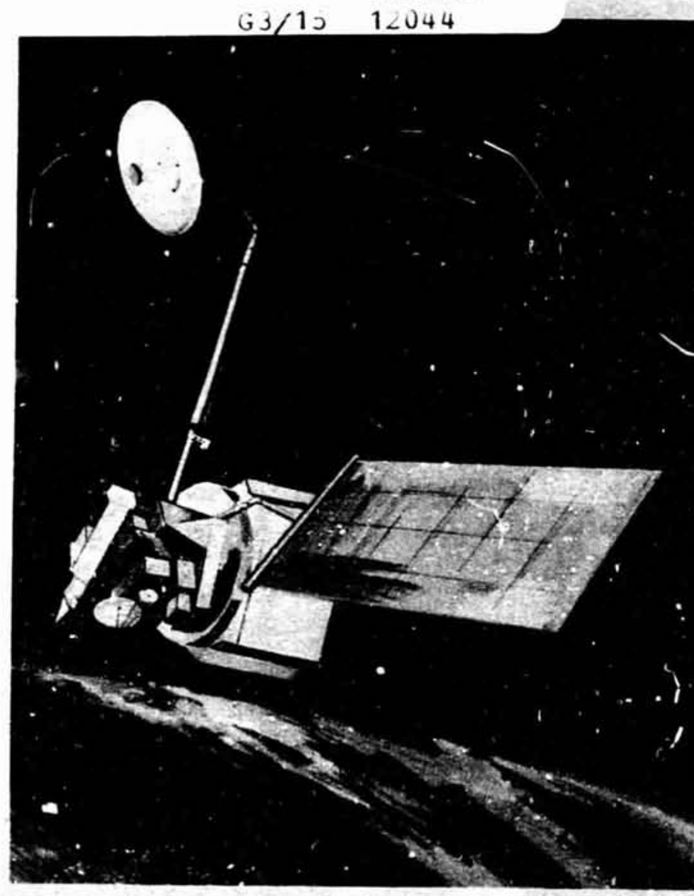

GRUMMAN

LANDSAT D INSTRUMENT MODULE STUDY

Prepared for

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SECTION 1

INTRODUCTION AND SUMMARY

1.1 INTRODUCTION

1.1.1 Scope of Study

Early in 1976, Grumman was awarded an engineering study by NASA/GSFC to develop alternate spacecraft Instrument Module configurations for advanced earth resource missions. The study was conducted in two phases: Phase 1 was an examination of a number of alternate configurations which satisfied the NASA design ground rules. This was followed by an evaluation and selection of the two designs which best suited the mission requirements. These two designs, one for each experiment complement, were then examined further in-depth in Phase II, Concept Validation.

The scope of the study included selection of the most promising candid[illegible] figurations and performance of the necessary design, analysis and modelling which would confirm the feasibility of these concepts. The solutions found in [illegible]s effort are neither unique nor absolutely optimum, but rather good solid design concepts upon which the Landsat Program can be planned.

1.1.2 Study Approach

The fundamental objective of this study was to establish viable spacecraft Instrument Module configurations which would support an earth resource data gathering mission using a Thematic Mapper sensor experiment designed by either Hughes or TRW. The differences in size of these two experiments necessitated the development of two different spacecraft configurations. Following the selection of the best-suited configurations, a validation phase of design, analysis and modelling was conducted to verify feasibility. The chosen designs were then used to formulate definition for a systems weight, a cost range for fabrication, and interface requirements for the Thematic Mapper (TM).

The study approaches used to develop and verify the Landsat Instrument Module Configurations are summarized in the study flow diagrams shown in Figures 1.1-1 and 1.1-2. Although the results of the study are embodied in the text of this entire report, a brief examination of this flow will give the reader an insight into the logic of the study plan.

Initially, a series of broad design criteria were set up to bound the configurations. Definitions of orbit, launch vehicle, payload capability, shroud envelopes, payload requirements, resupply, and GFE items were established. Using these criteria, a series of equipment arrangement and structural configurations were established. Parallel to this, a series of other efforts were conducted. These included: (1) an investigation of alternate appendage configurations; (2) an orbital flux analysis to be used in later thermal modeling; (3) definition of a preliminary loads environment: (4) establishment of gross thermal interfaces; and, (5) estimates of mass property characteristics for each alternate configuration. Using the results of the above efforts, each candidate was reviewed with respect to best satisfying the various design requirements. This qualitative evaluation resulted in the selection of two configurations, one for each Thematic Mapper for in-depth quantitative validation.

The two selected configurations H-1A (Hughes TM) and T-1A (TRW TM) were pursued with efforts in design, analysis and modeling. Two structures were developed, analyzed, sized, and modeled. Stress models yielded information on internal load distributions: dynamic models established launch and orbit modes and frequencies; and thermal models defined the heater power and insulation requirements. Orbital relationships were established between the solar array, TDRSS antenna, TDRSS satellite, the earth and the sun to optimize the position of the array and the antenna. Mechanical concepts were developed for appendage deployment mechanisms. Module exchange mechanism (MEM) adapters were developed which satisfied the in-orbit resupply requirements. Mass properties were defined for launch and in-orbit conditions. These efforts created an in-depth definition of the selected configuration and at the same time verified the choices made in the first phase of the study and provided definition for the Thematic Mapper interface with the Instrument Module. A program plan was developed which: (1) defined a work breakdown flow at three levels; (2) developed a schedule for a three flight program: and, (3) estimated costs for selected elements.

The remainder of this report presents the detailed results of this study effort.

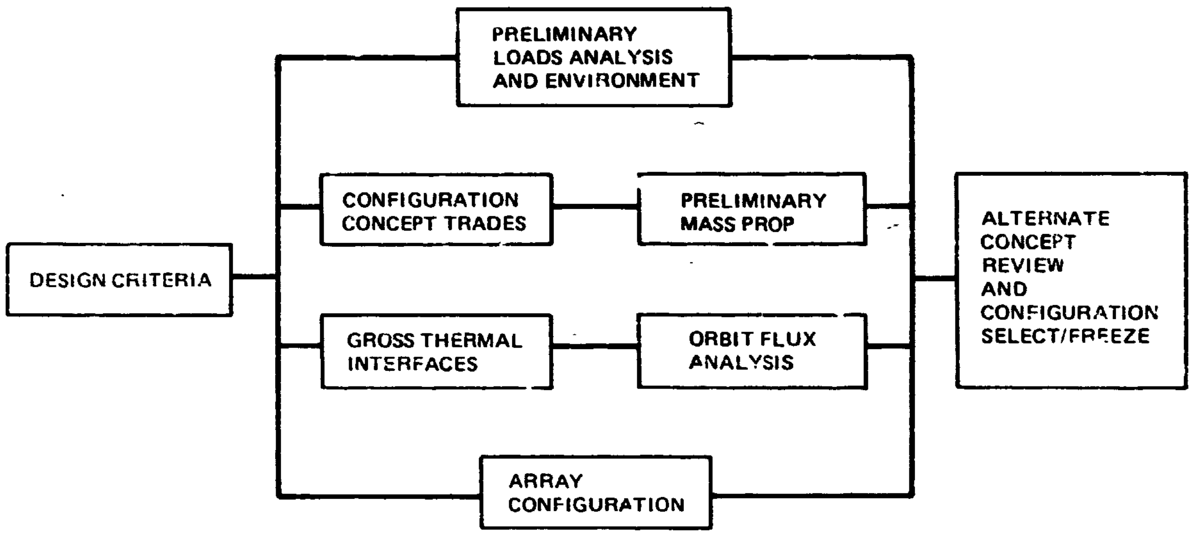

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FIGURE 1.1-1 CONCEPT SELECTION STUDY FLOW

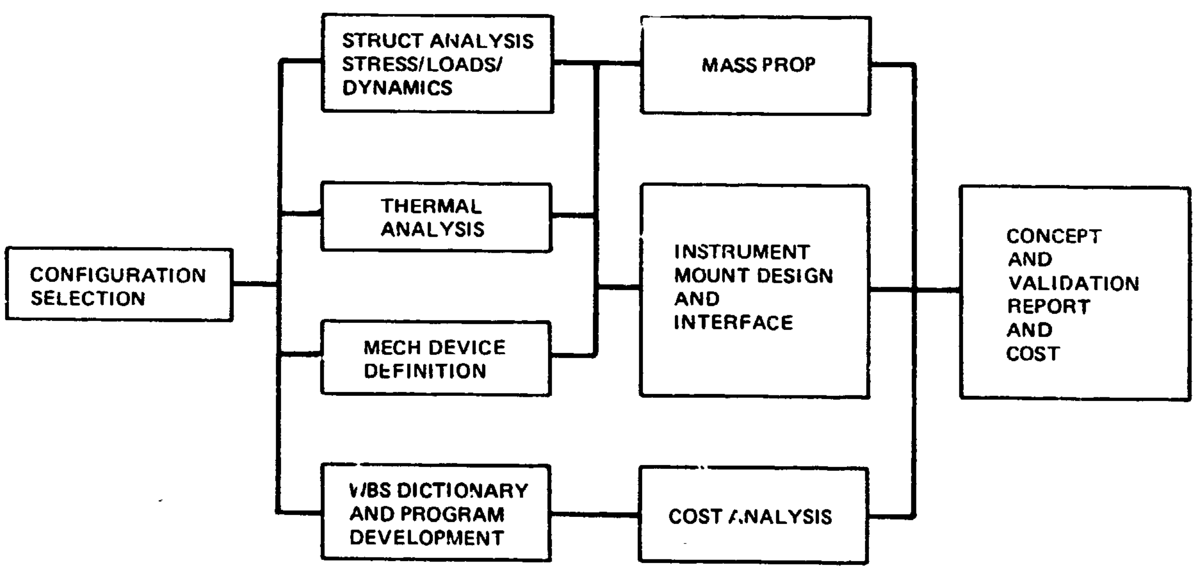

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FIGURE 1.1-2 CONCEPT VALIDATION STUDY FLOW

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1.2 SUMMARY

1.2.1 Major Criteria and Constraints

This section is a summary of the study results. Initially, a series of guiding criteria and constraints were developed to establish the bounds of the configuration development. The basic design task was to develop an instrument module which would contain: thematic mapper (TM) and multispectral sensor (MSS) instruments, a wide band module (WBM) and TDRS deployable antenna communication system, a deployable solar array (SA), and be compatible with the multi-mission modular spacecraft. A 705 km Sun-synchronous Polar Orbit with a 9:30 am descending node was dictated by mission requirements. Booster requirements, weight limitation and on-orbit refurbishment requirements were established. The major study criteria and constraints are:

- Major components - Thematic Mapper (both Hughes and TRW versions), Wideband Module, Multispectral Scanner, 1500 W Solar Array, 76-inch dia rigid TDRS antenna, multimission modular S/C
- 705 km sun synchronous orbit - 98.14° inclination with 9:30 a.m. descending node
- Delta 3910 launch from WTR
- Observatory maximum weight - 3,670 lb
- Delta shroud and shuttle payload compartment clearance envelopes
- Loads environment for Delta launch and shuttle retrieval
- Thermal isolation of experiments
- On-orbit experiment refurbish capability with MEMS
- Two S/C launch (1981) with refurbishment/resupply mission (1983).

In the initial investigations of compatible equipment arrangements, a number of strong configuration drivers were evident. Counterbalancing the spatial requirements of the Delta shroud, which tended to require high packaging density, the optical fields-of-view (FOVs) coupled with the orbital orientation requirements for earth-viewing tended to require a more spacious arrangement. Key configuration drivers that had to be satisfied for each option examined were:

- FOV for optics, radiators, and antennae
- Orbital orientation requirements of TM, WBM, MSS, TDRS antenna, solar array, and MMS
- Delta fairing dynamic envelope
- On-orbit experiment refurbishment requirement
- Structural continuity with MMS primary structure
- TM and MSS mounting plane requirements.

Initially, all spacecraft orientations were considered. Figure 1.2-1 illustrates the three basic positions: long axis in velocity direction (+X), long axis towards earth, (+Z) and the long axis normal to orbital plane (+Y). Due to the aforementioned constraints, certain orbital invariant direction relationships were shown to be evident. They were:

- Experiment fields-of-view (FOV): +Z
- TDRS antenna (anti-earth side): -Z
- Solar array (sun side): -Y
- Radiators (anti-sun side): +Y

1.2.2 Alternate Configurations

Eight different configurations were developed which, at least minimally, satisfied the basic requirements set forth. The spacecraft arrangements for these alternates are shown in Figure 1.2-2 in their compact stowed posture within the Delta shroud. Exo-structural definition was established and can be seen in Figure 1.2-3.

The following key comparative evaluator items were used to determine the best configurations:

- Structural efficiency
- Low weight
- Center of gravity
- Portability and accessibility
- Commonality

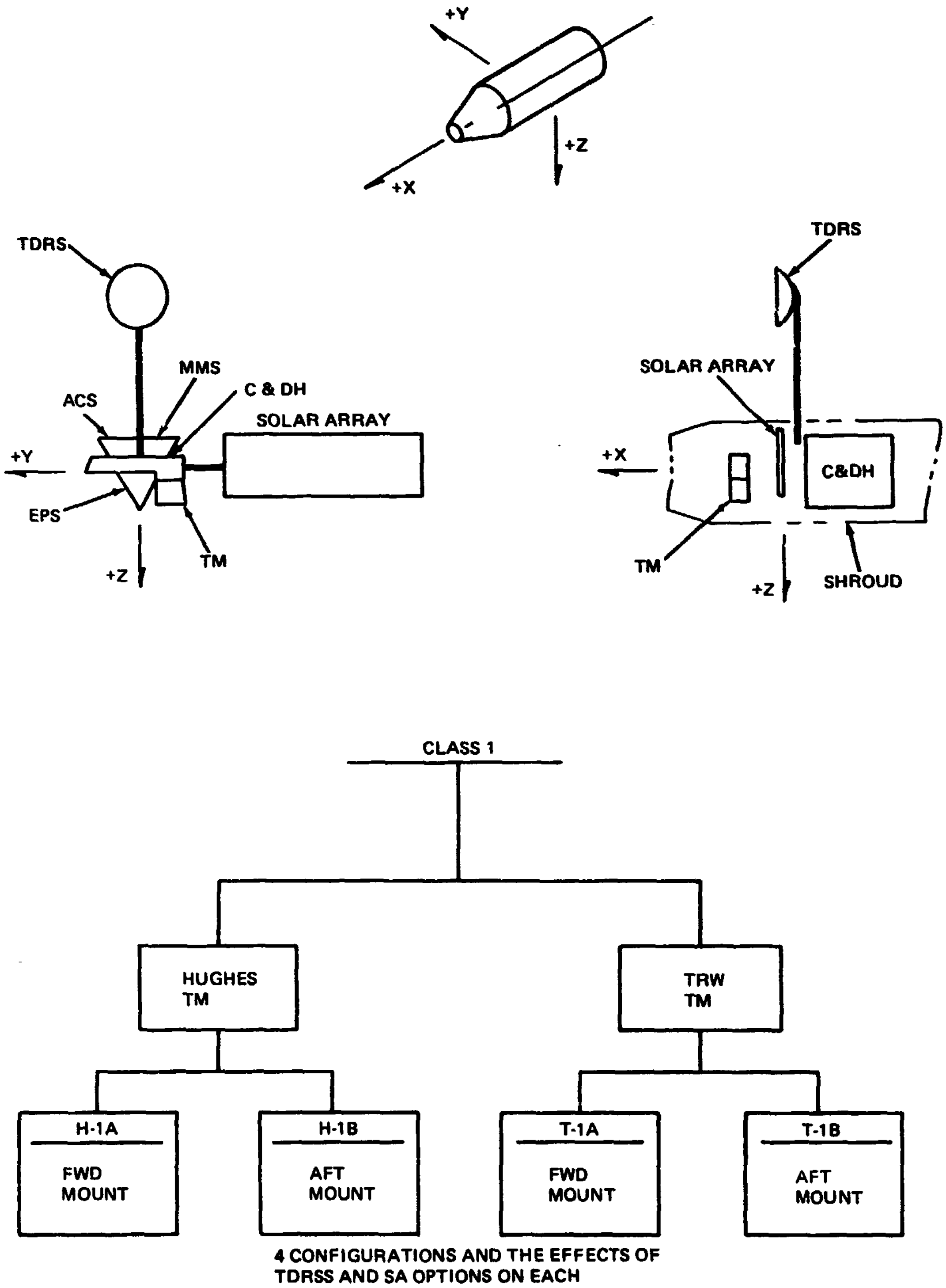

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FIGURE 1.2-1 ORIENTATION CLASSES OF CONFIGURATION (SHEET 1 of 3)

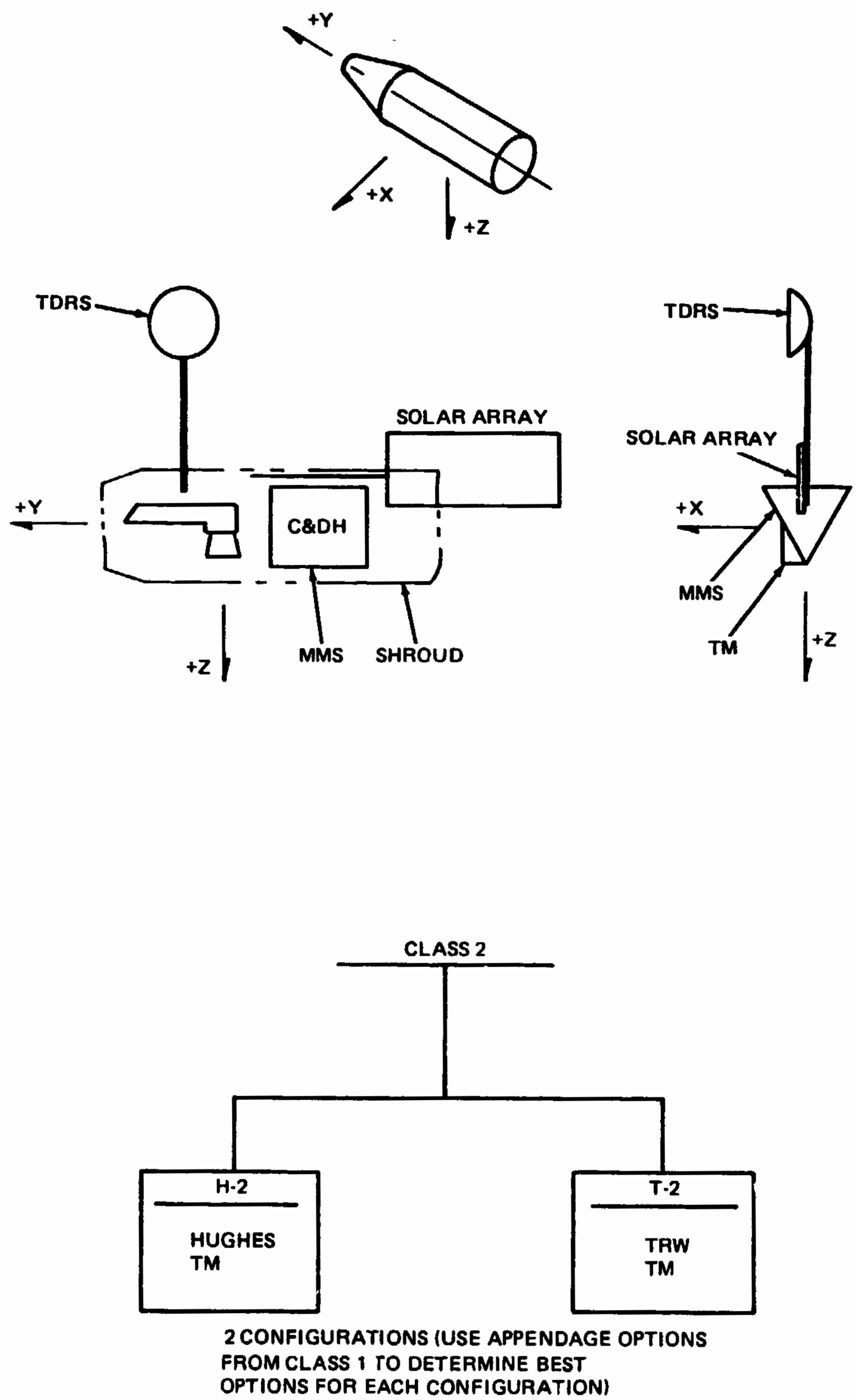

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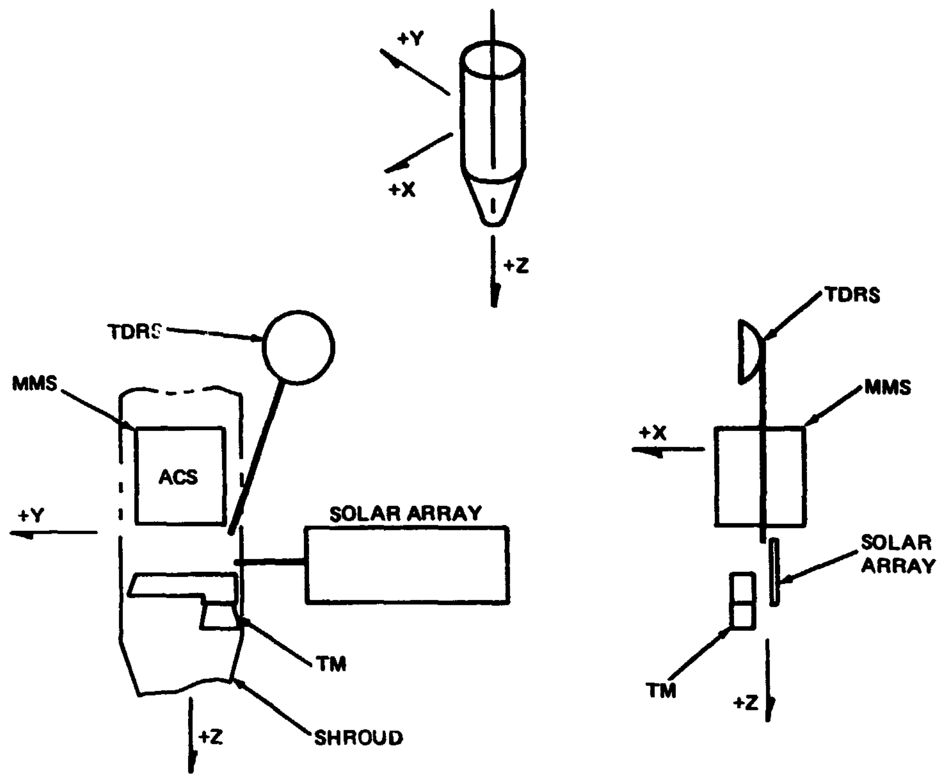
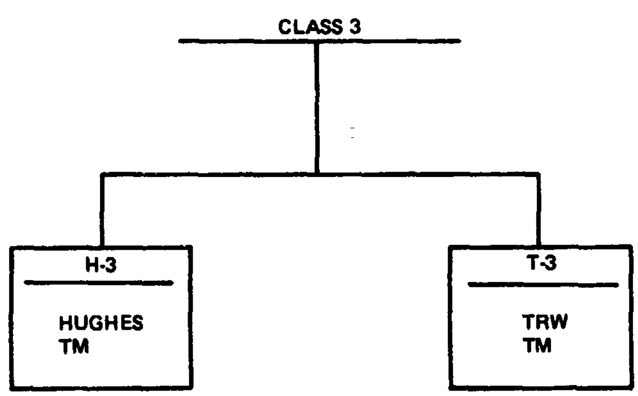
FIGURE 1.2-1 ORIENTATION CLASSES OF CONFIGURATION (SHEET 2 of 3)





2 CONFIGURATIONS (USE APPENDAGE OPTIONS
FROM CLASS 1 TO DETERMINE
BEST OPTIONS FOR EACH
CONFIGURATION)

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FIGURE 1.2-1 ORIENTATION CLASSES OF CONFIGURATION (SHEET 3 of 3)

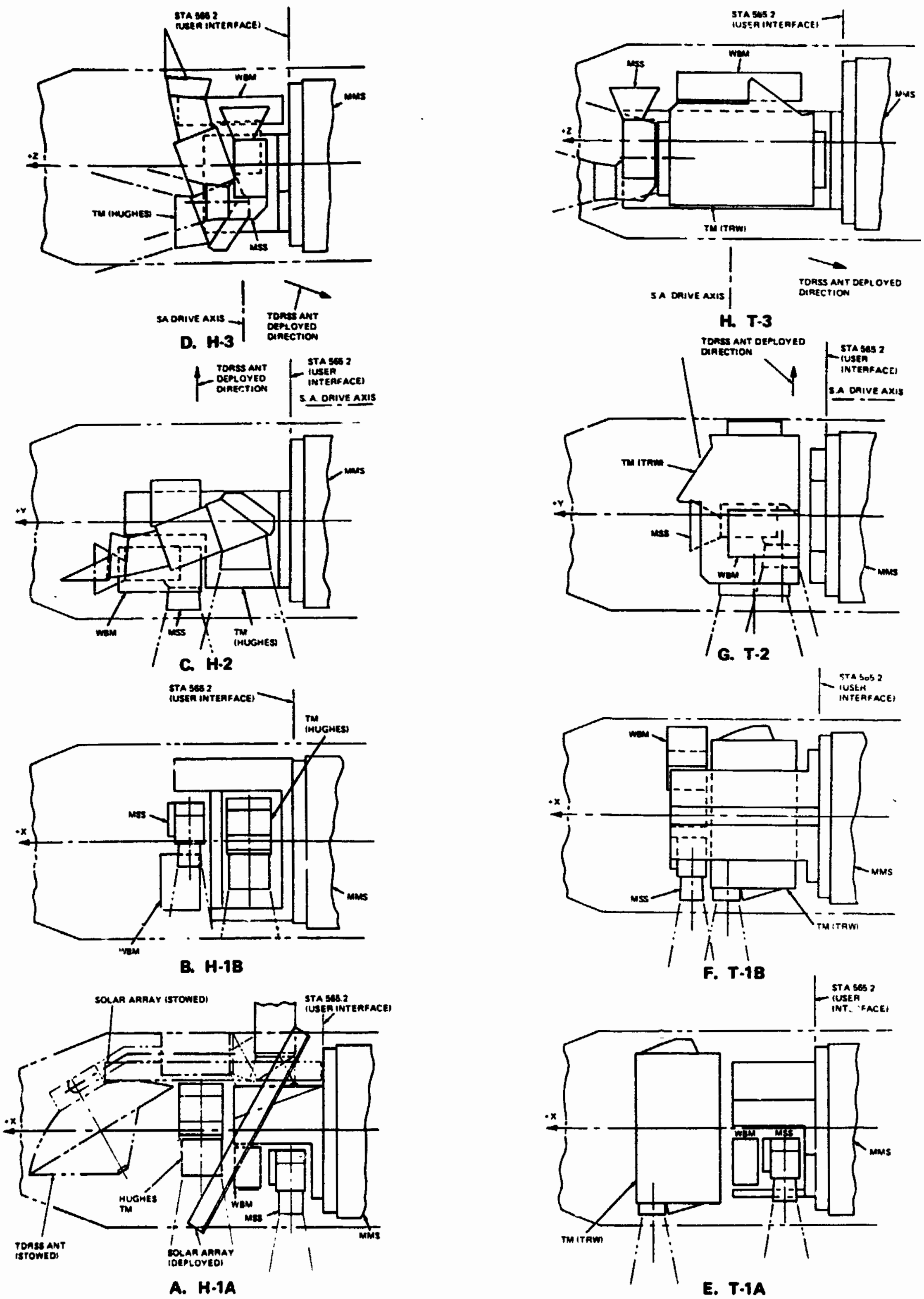

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FIGURE 1.2-2 ALTERNATE FIGURATIONS

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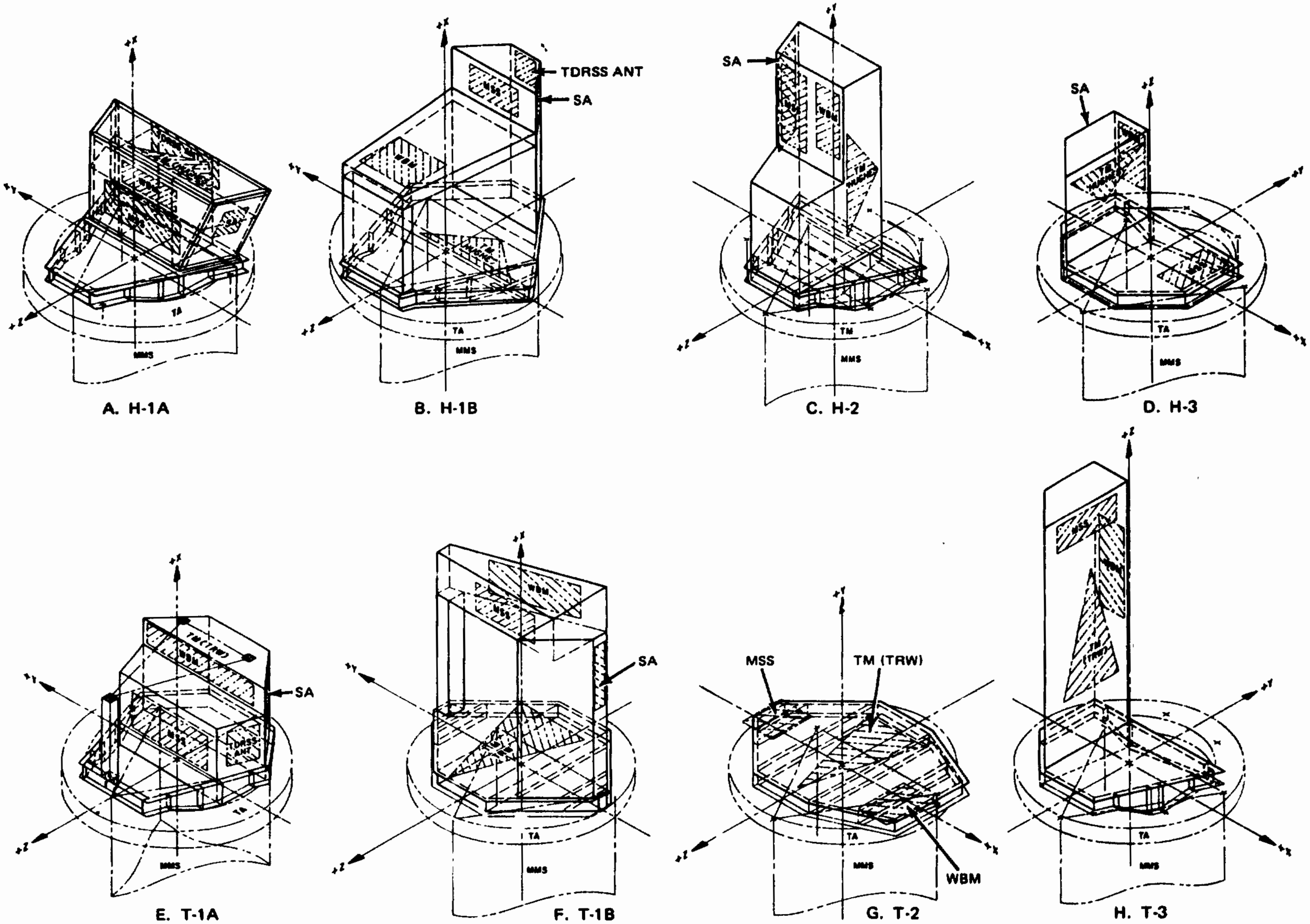

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FIGURE 1.2-3 ALTERNATE EXO-STRUCTURE

- MMS continuity
- TM mount
- Base bulkhead depth
- Thermal surfaces
- MMS orientation
- Orbit reorientation
- FOV
- Radiator view
- Packaging efficiency
- Structural flexibility
- Stowed appendages
- Antenna/array deployment

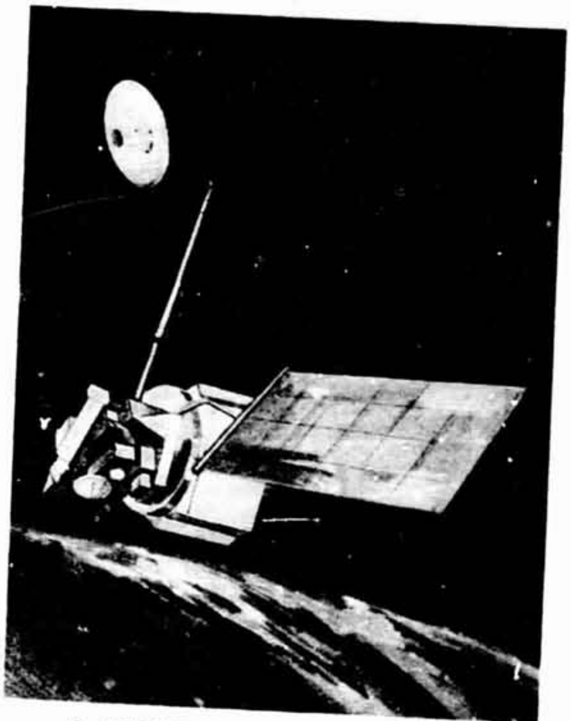
The final selected configurations H-1A and T-1A were most compatible with these requirements.

1.2.3 Verification Phase

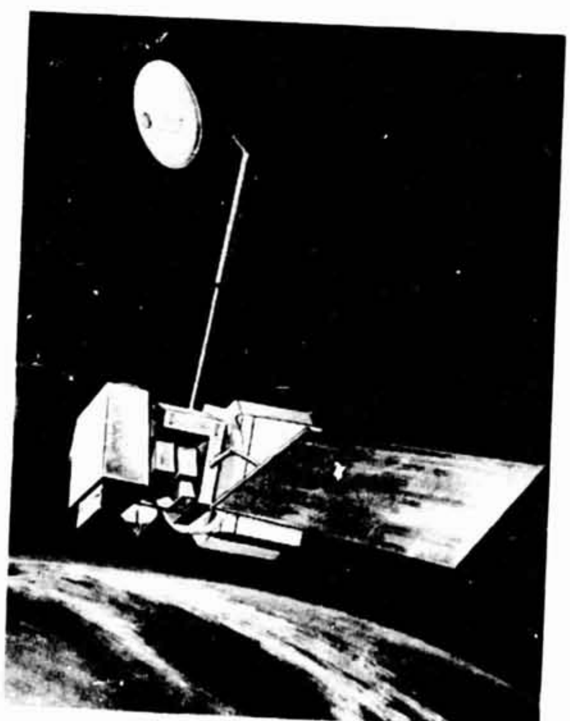
Subsequent to the selection, a verification phase was initiated to technically justify the preliminary choices. A structural design and stress analysis effort established a fully sized primary structure of aluminum for each of the thematic mappers. Structural and dynamic math models were analyzed to determine that the fundamental modes, frequencies, and excursions remained within the envelope conditions. Mechanical deployment configurations were developed for the TDRSS antenna, the solar array, and in-orbit refurbishment capability. An effective thermal configuration was developed which used a nominal amount of heater power, coupled with multilayered insulation and the use of titanium experiment mounts. Figures 1.2-4 through 1.2-7 illustrate the chosen configurations.

Detailed definition of the spacecraft configurations are presented in Sections 2 and 3 of this report. The key features of the designs selected are the following:

- Efficient structure - weight, portability, accessibility, stiffness
- Base Bulkhead provides good continuity to MMS -
- Short TDRS antenna requirement - allows simple rigid link deployment
- Vertical box struct - interior space available for secondary equipment



A. HUGHES TM–CONFIGURATION H-1A



B. TRW TM–CONFIGURATION T-1A

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FIGURE 1.2-4 ORBITAL ISOMETRIC

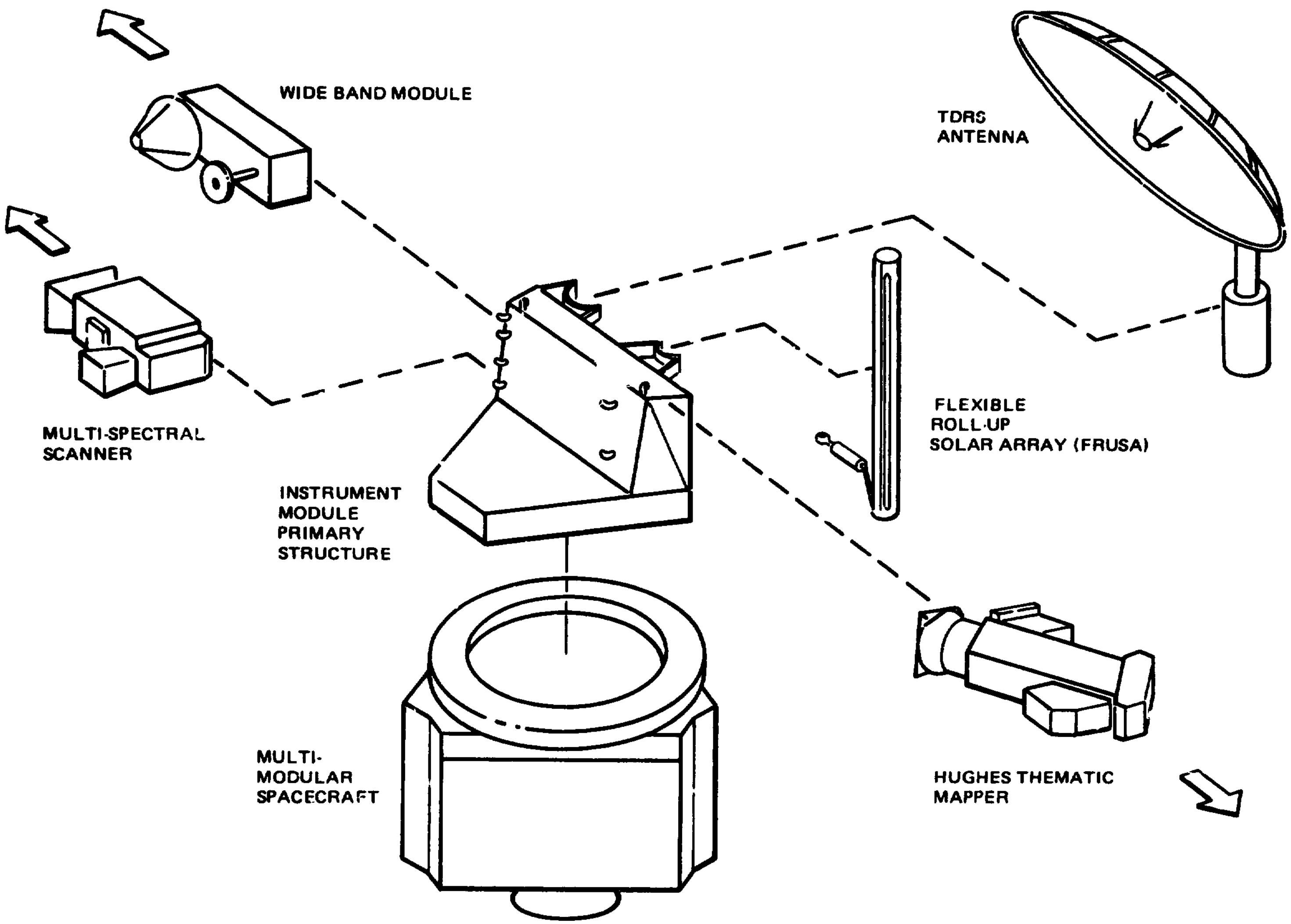

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FIGURE 1.2-5 EXPLODED VIEW – H-1A

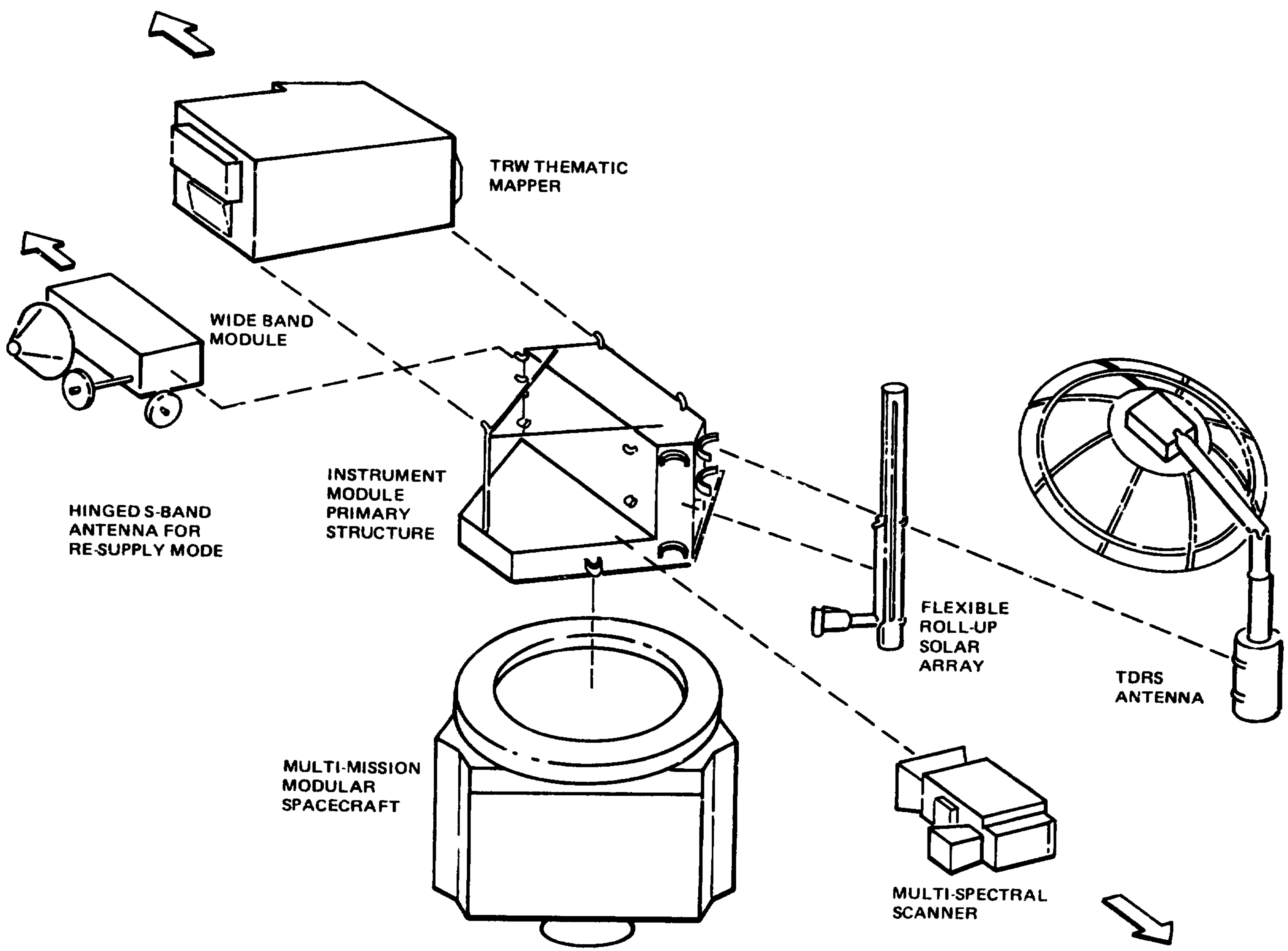

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FIGURE 1.2-6 EXPLODED VIEW – T-1A

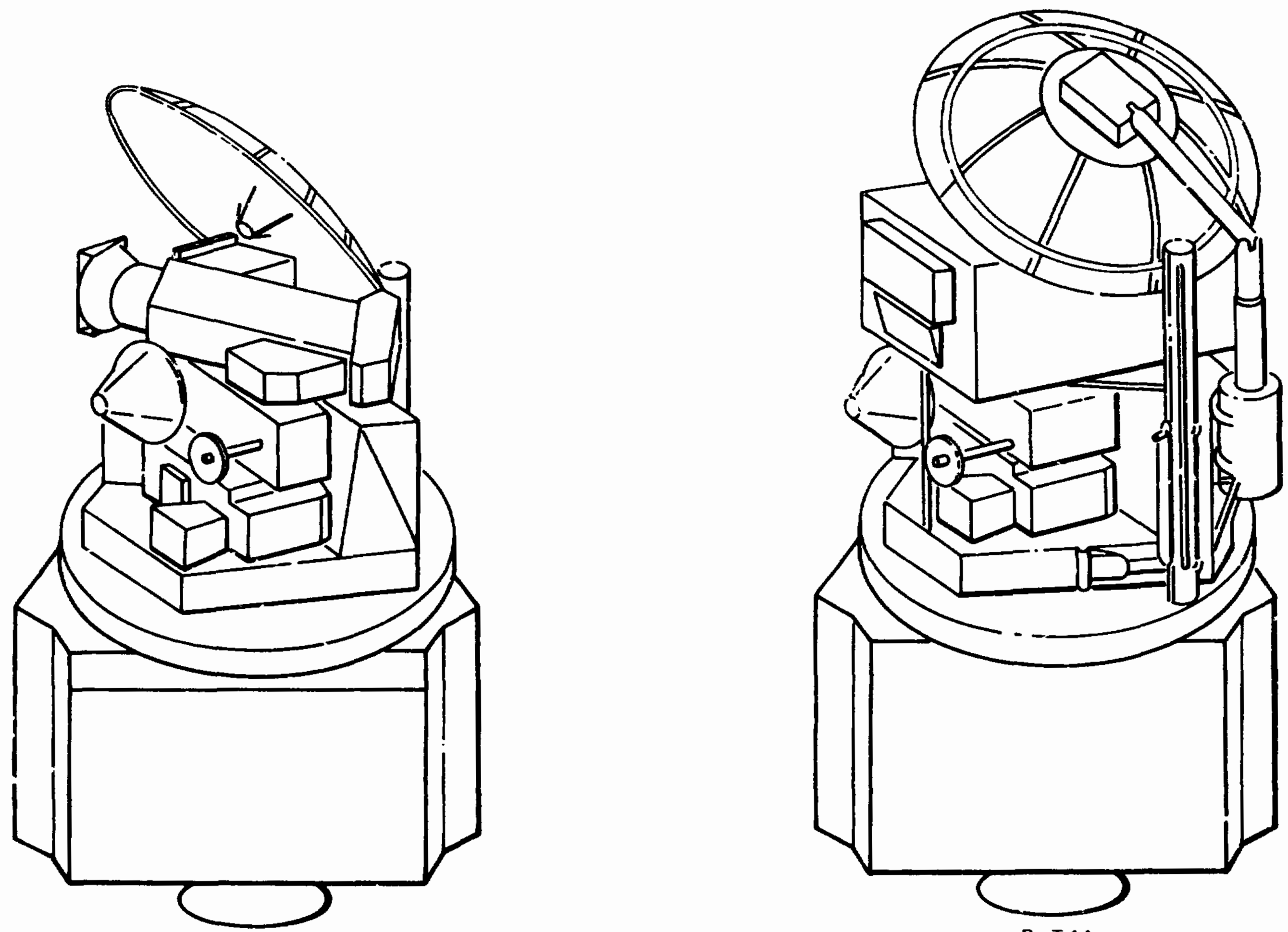

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FIGURE 1.2-7 LAUNCH CONFIGUR/.TIONS

- Fundamental launch frequency - 14 Hz lateral, 32 Hz longitudinal
- Fundamental orbit frequency - 0.33 Hz FRUSA bending
- Aluminum structure with titanium fittings - sheet/stringer construction
- Simple thermal system - multi-layer ins., heaters, titanium fittings
- Nominal Htr pwr - 30-40 watts
- Analysis models - structural/dynamic/thermal
- Compact stowed arrangement.

1.2.4 Description of Selected Configurations

Although the basic differences between the two thematic mappers dictated designs with significantly different dimensional requirements, there is strong commonality between the features of the two configurations. In fact, a single brief generalized description of the Landsat Instrument Module design is sufficient for both.

The spacecraft has a gross weight of approximately 3,670 lb which is packaged in its stowed configuration within an 84-in. dia Delta shroud envelope. The major components are:

- Multi-mission modular lower spacecraft (MMS) - provides housekeeping and propulsion services
- Structure - supports the equipment and is designed for Delta launch loads
- Experiment Sensors - includes a Thematic Mapper (either TRW or Hughes design) and a multi-spectral scanner
- Wide band module - has Ku and S band antennae directly mounted to the module front face
- Thermal System - provides thermal isolation and maintains all temperatures within operating limits
- Transition Ring - provides load paths from the base of the Instrument Module to the MMS and has adaptations for shuttle use
- Flexible Roll-Up Solar Array (FRUSA) - provides sun-synchronous power

- 76-inch articulating dish antenna, boom mounted - provides a continuous data link to the Telemetry Data Relay Satellite System (TDRSS)
- Mechanical Systems - provides orbital deployment capability for the TDRSS Antenna and the FRUSA array as well as mounting provisions for in-orbit refurbishment.

The structure is an L-shaped box-like aluminum sheet metal design with titanium equipment mounting provisions which permit in-orbit refurbishment and replacement. The Thematic Mappers are positioned at the forward end of the Instrument Module structure. In the Hughes configuration the three sensor pick-up points are mounted to titanium fittings on the X = 45 bulkhead. The aft two pickup points for the TRW sensor are similarly mounted on the X = 51 bulkhead. However, the forward pickup point is supported by means of a vertical column connected directly to the aft bulkhead. The forward position of the Thematic Mapper was dictated by volumetric considerations. The large Thematic Mapper cross-sections were required to be stowed within a very restrictive 84 inch diameter shroud envelope. An aft position for the mappers would block the space required for an effective structure.

The equipment is mounted on the top and forward faces of the structure to permit unblocked earth viewing (+Z axis) for the optics and communications system, and access for radiation on the anti-sun side. The equipment is arranged compactly, as required by the shroud confines, but with sufficient spacing to permit uninhibited fields-of-view and direct access for in-orbit refurbishment. The FRUSA array and the TDRSS antenna are stowed compactly alongside and above the structure in proximity of their deployed quadrant positions. The thermal system consists of: multi-layered insulation wrapped around the structure and each individual equipment; heaters controlled by solid-state thermostats mounted in the structure; and, titanium equipment mounting fittings to minimize conductive heat paths. Sections 2 and 3 discuss each design configuration in detail.

SECTION 2

SELECTION PHASE

2.1 INTRODUCTION

Recently, Grumman was awarded a design study by NASA/GSFC to develop alternate spacecraft configurations for the Instrument Module of an Advanced Sun-synchronous, polar orbit earth resource satellite designated Landsat D. The results of this study are to be used by GSFC in the planning and costing of a major new satellite program in the 1978-1981 time period.

A two-phase study program has been conducted. Phase I deals with the examination of a number of configurations and the selection of two (one for each of two instrument complements) arrangements for further in-depth study. Phase II is Concept Validation and Costing.

The basic task of the selection phase was to develop a series of alternate arrangements which satisfied the major criteria and constraints and which was volumetrically compatible within the tight bounds of the Delta shroud in the stowed position without violating orbital requirements. Provisions were required for: fields-of-view (FOV's) of experiments, antenna clear lines of sight, erectable array and antenna, access for orbital modular exchange, radiation views to cold space, continuous structure, compatibility with MMS, as well as other more specific requirements.

Having established these alternate configurations, each alternate was then subjected to a set of comparative evaluation criteria related to: structure, thermal, volumetrics, and general configuration suitability. The best suited configuration for both the Hughes and TRW Thematic Mappers were then selected for an in-depth verification phase consisting of design, analysis and math modeling investigations. The remainder of this section deals with the establishment of the selected configurations.

2.2 CRITERIA

In order to establish design criteria for the Landsat Instrument Module, a set of guidelines were established for configuration development, structural and thermal environment, and spatial constraints.

Figure 2.2-1 lists the key configuration criteria. These criteria fell into three classifications. First, hard criteria such as fields-of-view were invariant requirements. Second, soft criteria, such as equipment weights, were only initial estimates and subject to possible variations as the program progressed. Third, a TBD classification such as alignment, where specific values were not established - although it was recognized that these open items were to be considered. In the case of alignment, environmentally caused structural distortions were to be minimized.

Table 2.2-1 is a summary of the environmental criteria that were used to design the aluminum instrument module structure. These values are worst case envelope derivations of the Delta booster and shuttle environments.

The following is a list of Thermal constraints and interface requirements for design:

- Thermally isolate components from module structure. For instruments and wideband module, conductive coupling for each to be less than 2 watts/°C
- Nominal operating temperature for instruments and wideband module = 20°C
- MMS structure to be at same temperature as instrument module structure (no heat transfer across this interface)
- Maintain instrument module structure temperature between 0°C and 40°C (20°C nominal)
- Design orbit is sun-synchronous; 705 km altitude; 0930 DN.

The key requirement is that the structure and each piece of equipment remains thermally isolated from each other and heat transfer is kept to an absolute minimum.

This criterion will provide a spacecraft with a maximum thermal stability.

FIGURE 2.2-1 DESIGN CRITERIA

ORBIT

705 km (380.6 N Mi), 9:30 am descending node, sun-synchronous, 98.14° inclination.

LAUNCH VEHICLE

Delta 3910 (WTR launch site)

PAYLOAD CAPABILITY

3670 lb

SHROUD

96 in. dia (clearance envelope of 86 in. dia)

PAYLOAD

- HUGHES THEMATIC MAPPER (Dwg PL1162 M102B)

  Orbital Orientation – Optics on +Z axis
  Radiator on +Y axis
  FOV – Optics ±15° in YZ plane about Z axis
  10° in XZ plane about Z axis
  Radiator ±80° in XY plane about Y axis
  +25.7°, -90° in YZ plane about Y axis
  Weight – 250 lb
  Duty cycle – Operating only over sunlit land masses and continental shelves
  Alignment – Inst'l ±0.2° of reference axis
  Launch ±0.1° shift
  On orbit – TBD
  Structural Provisions – YZ mounting plane

- TRW/PE THEMATIC MAPPER (Dwg AD 78-33)

  Orbital Orientation – Optics on +Z axis
  Radiator on +Y axis
  FOV – Optics ±15° in YZ plane about Z axis
  ±10° in XZ plane about Z axis
  Radiator ±50.5° in XY plane about Y axis
  +15.5°, -85.5° in YZ plane about Y axis
  Weight – 650 lb
  Duty cycle – Operating only over sunlit land masses and continental shelves
  Alignment – Inst'l ±0.2° of reference axis
  Launch ±0.1° shift
  On-orbit – TBD
  Structural Provisions – YZ mounting plane

- WIDE BAND MODULE

  Alignment – Inst'l – TBD
  Launch – TBD
  On-Orbit – TBD
  Structural Provisions – TBD

- TRACKING DATA RELAY SATELLITE ANTENNA

  Configuration – – 76 in. dia rigid antenna Z axis gimbal
  FOV – to sight two geostationary satellites at longitude 45° W and 168° W at 0° latitude
  Duty Cycle – Transmit only over sunlit portion of earth
  Alignment – TBD
  Structural Provisions – TBD

- SOLAR ARRAY

  Size 1500 watt array (beginning of life)
  Orbital orientation – -Y axis (1 axis gimbal)
  Wt – 150 lb
  Type – FRUSA

RESUPPLY

Resupply TM, MSS, & WBM
Retract TDRS Antenna and Solar Array for Resupply of either TM, MSS, WBM or Instrument Module/MMS

MULTI-MISSION MODULAR S/C

Transition Ring at sta 565.2 is Instrument Module Interface
Long axis of MMS coincides with velocity vector
Power Module faces +Y side or orbit plane
Stiffness matrix for MMS-TBD
Load Capability of MMS-TBD

2473-73

TABLE 2.2-1 STRUCTURAL ENVIRONMENT CRITERIA

	Ultimate Load Factor*		
	X	Y	Z
Delta MECO	17.6 G	1.5	–
STS Crash	–9.5	–	–
Delta Liftoff	3.6	3.0	–
STS Entry	±.38	±.75	–4.5

Ultimate Sinusoidal Vibration*			
Thrust Axis		Lateral Axis	
Frequency, Hz	Level G	Frequency, Hz	Level G
5-6.2	0.75 inch DA	5-100	.8 inch DA at 5 Hz 1.05
6.2-15	1.5	–	–
15-21	4.5	–	–
21-100	1.5	–	–

Desirable Fundamental Frequencies
• 35 hz in the thrust axis • 15 hz in the lateral axis

*Using factor ultimate load/limit load of 1.5

2473-14T

Figure 2.2-2 illustrates the volumetric problem involving large thematic mapper cross-sections inscribed in a very restrictive 84 in. dia shroud envelope. The shaded area indicates where instrument module structure must be placed in order not to infringe upon the experiments' airspace. The squeezed volume problem substantially influences all viable candidate configurations as will be seen in later discussions.

2.3 CLASSIFICATIONS

In the initial search for the most viable spacecraft configurations, all orientations were considered. This resulted in three basic classifications:

- Class One - the long axis in the velocity direction
- Class Two - the long axis normal to the plane of orbit
- Class Three - the long axis pointed towards the earth.

Refer to Figure 1.2-1 for these orientations. Due to certain invariant orbital requirements, some of the equipment must remain fixed and independent of spacecraft orientation. These items are:

- TDRSS Antenna - This articulating antenna must remain "high" above the spacecraft in the -Z direction in order to provide maximum unblocked viewing of the TDRSS satellite
- Solar Array - The rotating sun synchronous array is fixed in the -Y sun direction
- Radiator Coolers - For maximum cooling they are positioned on the +Y anti-sun side
- Experiment Viewing - FOV's must be directed towards earth nadir in +Z direction.

The MMS preferred direction, Class one, was a consideration but was not considered to be an invariant.

The investigation into the three classes of orientations gave rise to eight discrete configuration candidates. The major differentiations of each option were: (1) the spacecraft orientation, (2) The Thematic Mapper (all other equipment remained invariant) (3) The positions of each item of equipment - in particular, the

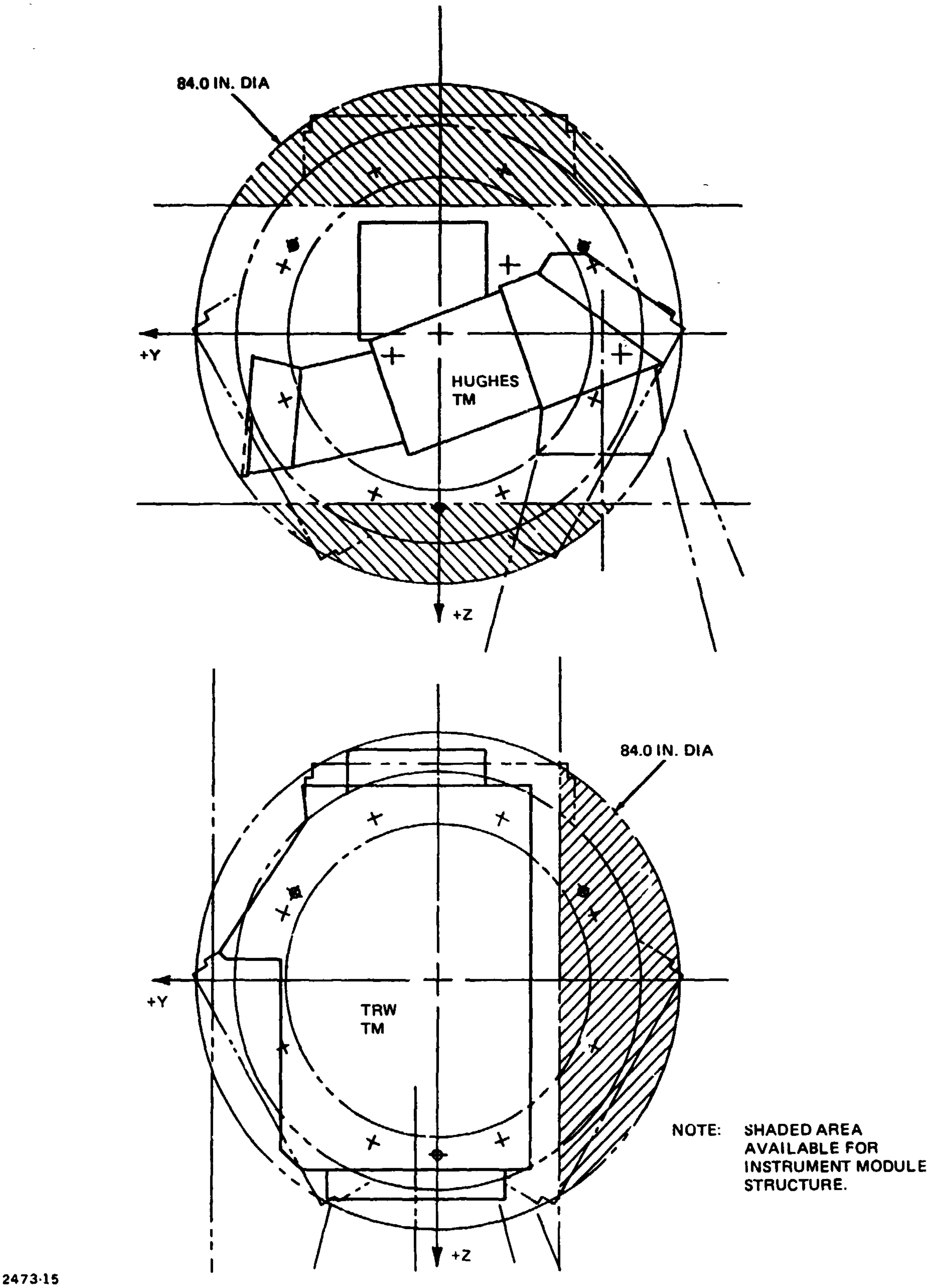

FIGURE 2.2-2 SHROUD VOLUMETRICS

Thematic Mapper. In addition to the eight configuration options, various arrangements were being considered for the solar array and the TDRSS antenna.

2.4 ALTERNATE CONFIGURATIONS

This section contains the eight major alternate configurations examined during Phase I. Equipment configurations, structural patterns and exostructural isometrics are incluc. .. It should be noted from the sketches that the major configuration drivers were: tight Delta shroud volume, large TDRSS antenna, large thematic mapper volume, fields-of-view for experimentation, communication, radiation, structural continuity, mechanism complexity, and appendage FOV interrelationships.

Figures 2.4-1 through 2.4-16 illustrate the stowed spacecraft equipment arrangement options and the resulting exo-structural shapes. In deriving the potential arrangements, geometric considerations were of paramount importance. A more complete analysis of these configurations will be shown in Subsection 2.7. Weights, centers of gravity (cg's), and moments-of-inertia have been calculated for each of the alternate configurations studied in Phase I. The weights used were initial study inputs with the particular geometry of the alternate configurations applied. Calculations were made for both stowed and deployed configurations. Near the end of Phase II a more accurate determination of mass properties of the two matured selected configurations were made. However, the preliminary results did not indicate a major weight problem.

Associated with these configurations, mass property calculations were made. These preliminary weights must be considered as estimates since they were made based upon early concept definition. They are listed in Table 2.4-1.

Table 2.4-2 shows the weight, centers of gravity, and moment-of-inertia associated with the eight configurations in the stowed position. Table 2.4-3 shows the same properties for several alternate deployed cases. These figures illustrate the influence of varying the solar array position as well as varying the antenna mast height. The values are for the instrument module only.

Table 2.4-4 shows a worst case compilation of orbital mass properties which includes all elements of the observatory. The most significant data here involve the shift of the center of gravity in the YZ plane from the symmetrical longitudinal axis.

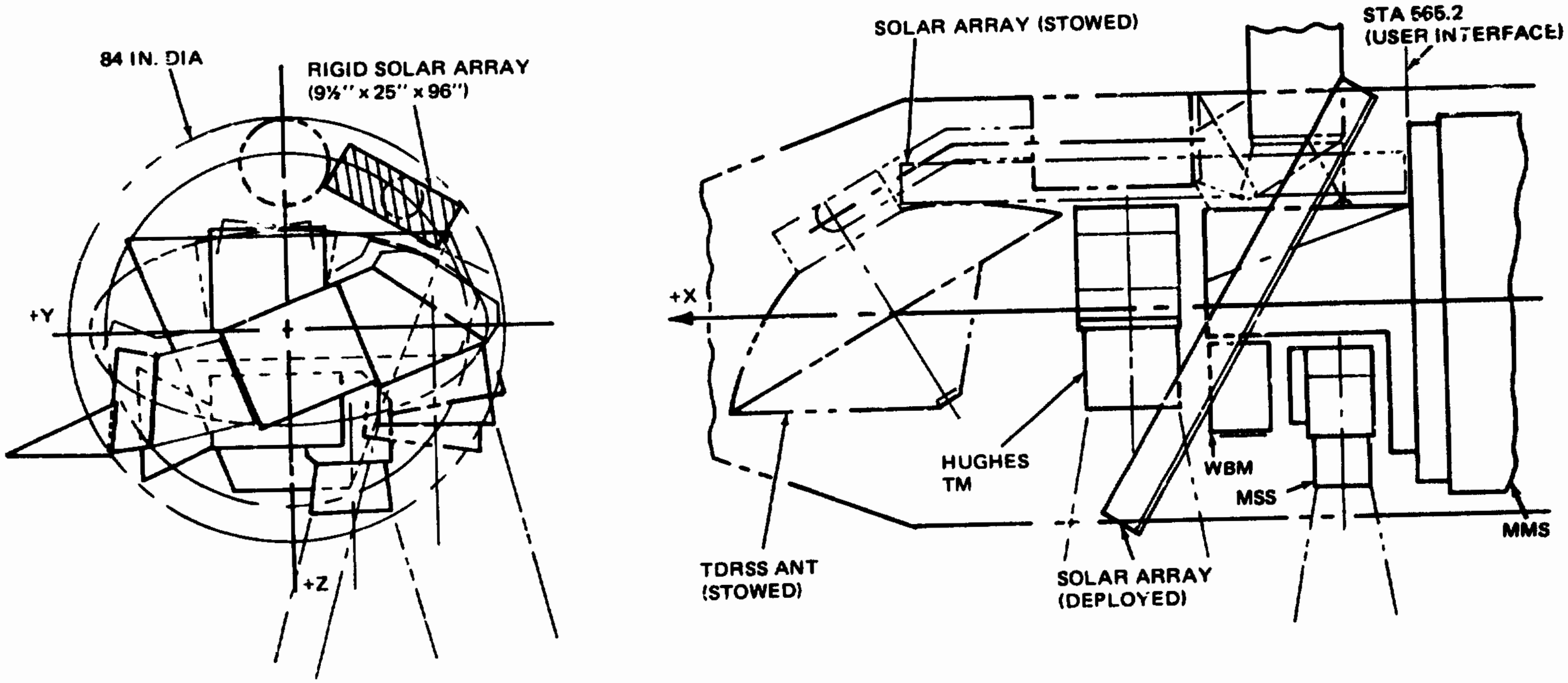

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FIGURE 2.4-1 CONFIGURATION H-1A

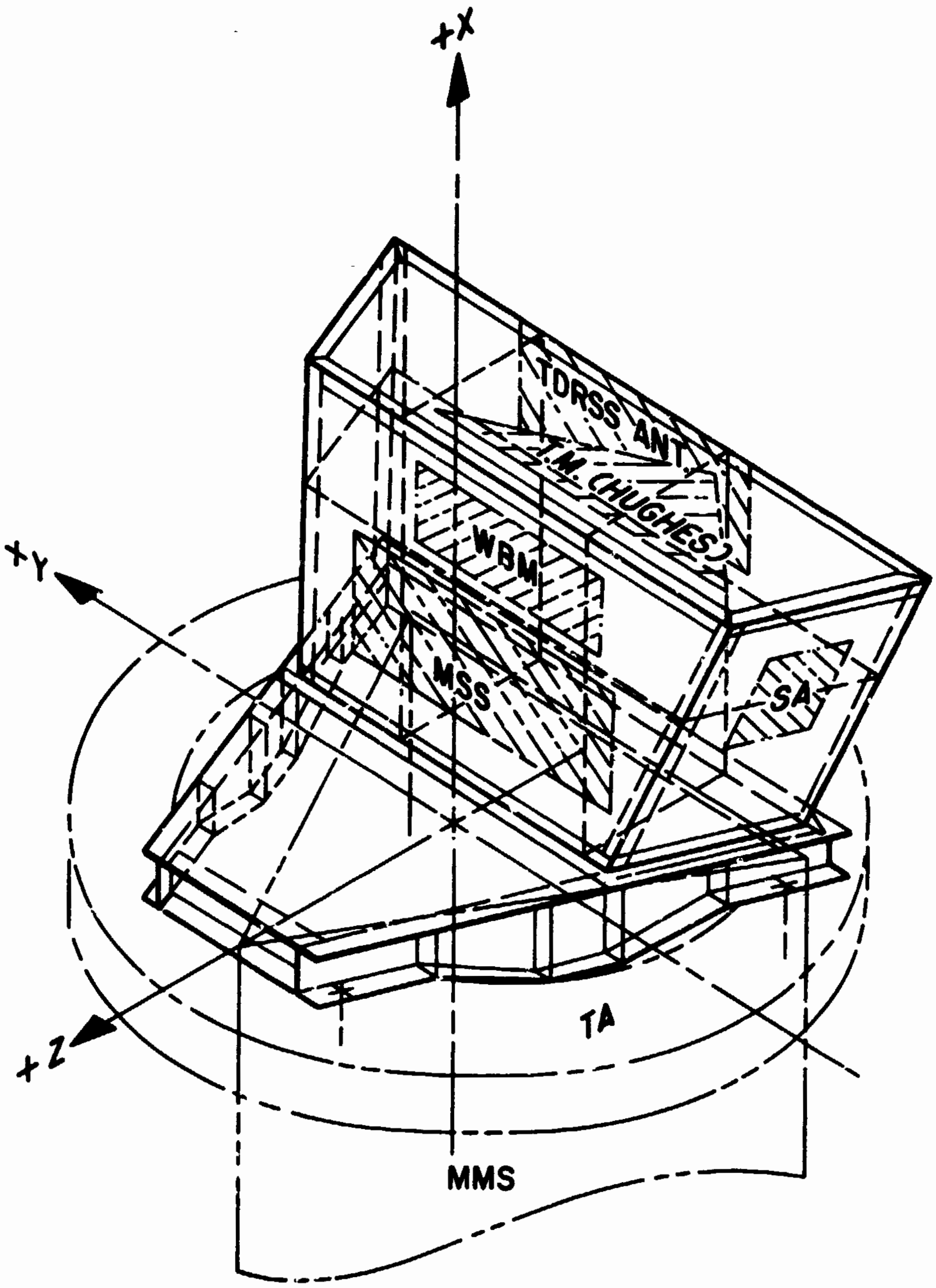

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FIGURE 2.4-2 STRUCTURE H-1A

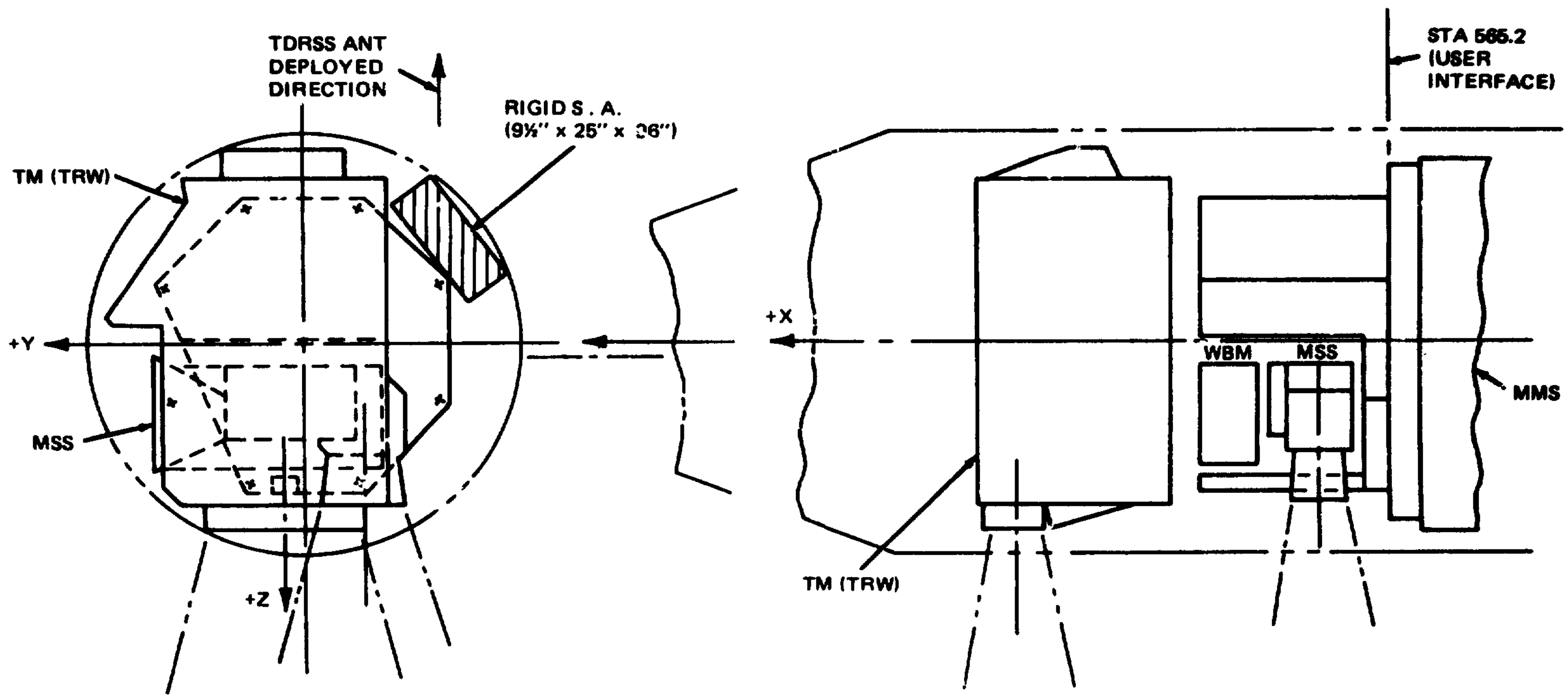

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FIGURE 2.4-3 CONFIGURATION T-1A

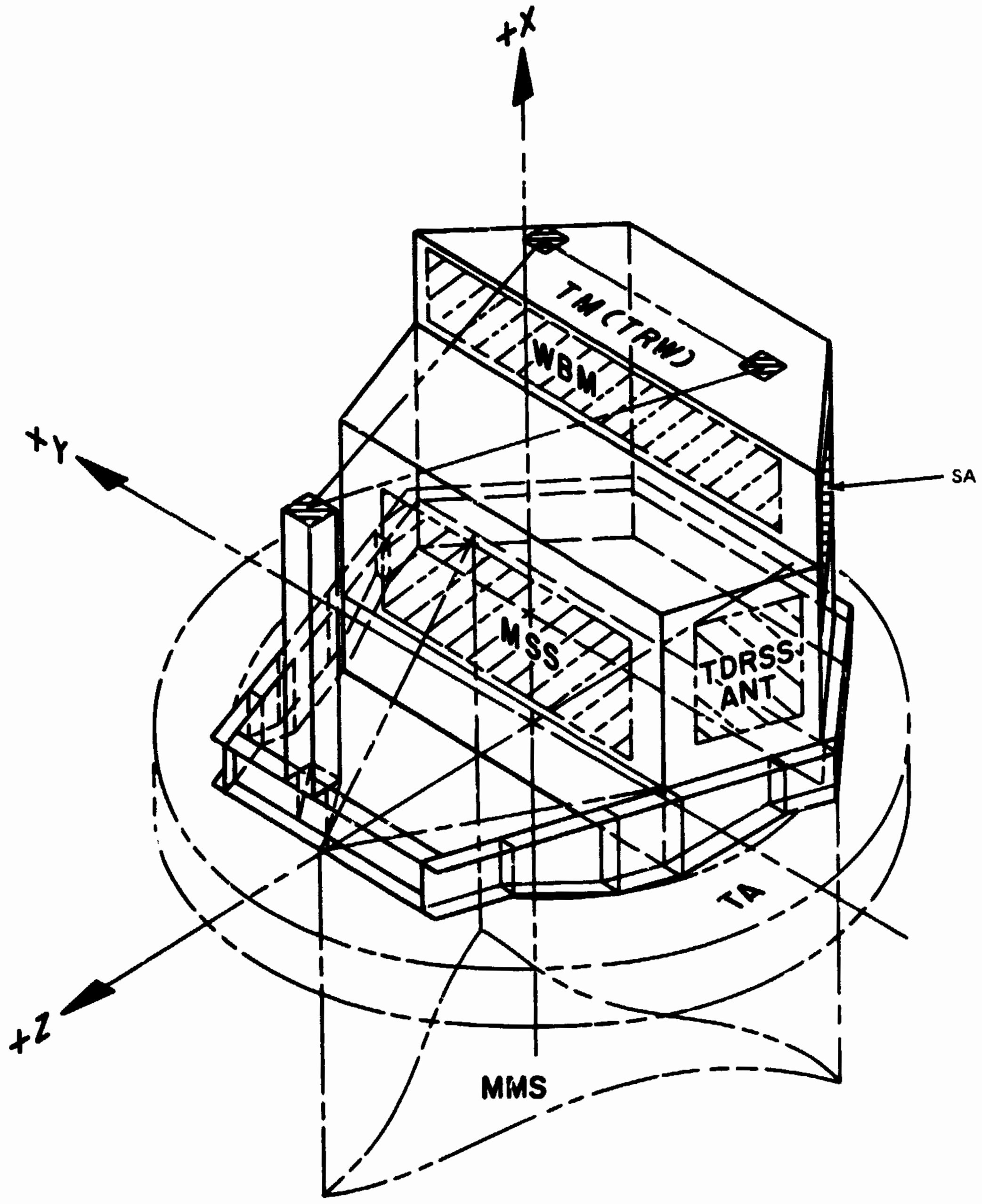

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FIGURE 2.4-4 STRUCTURE T-1A

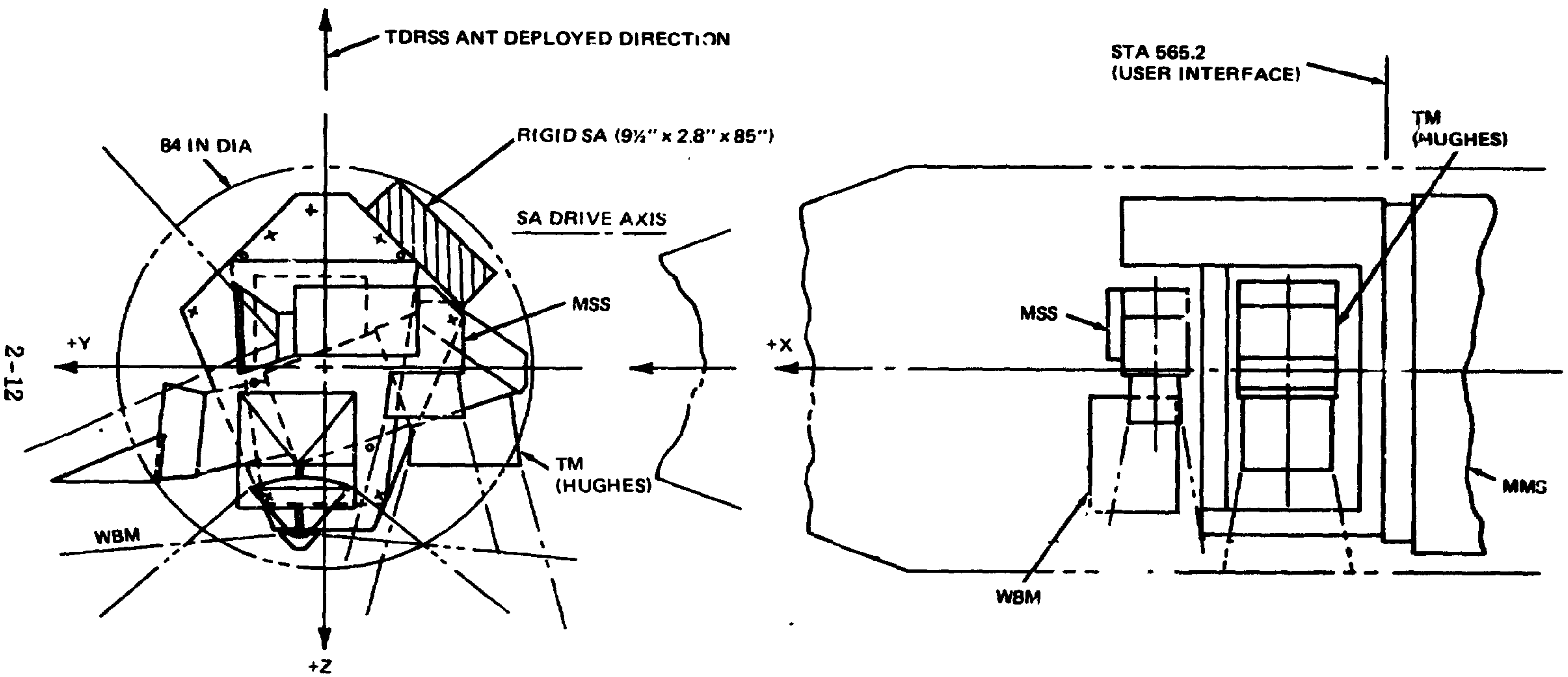

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FIGURE 2.4-5 CONFIGURATION H-1B

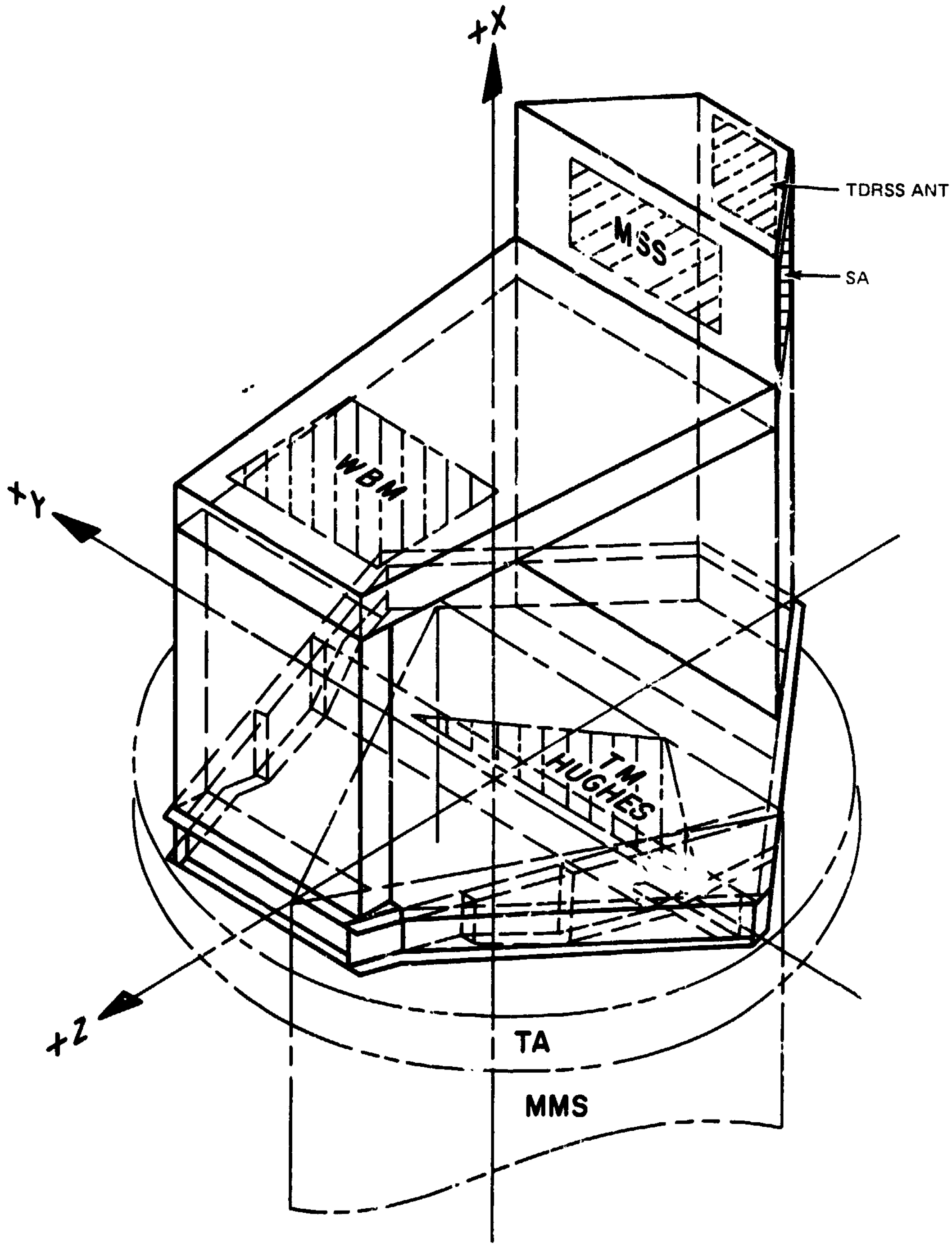

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FIGURE 2.4-6 STRUCTURE H-1B

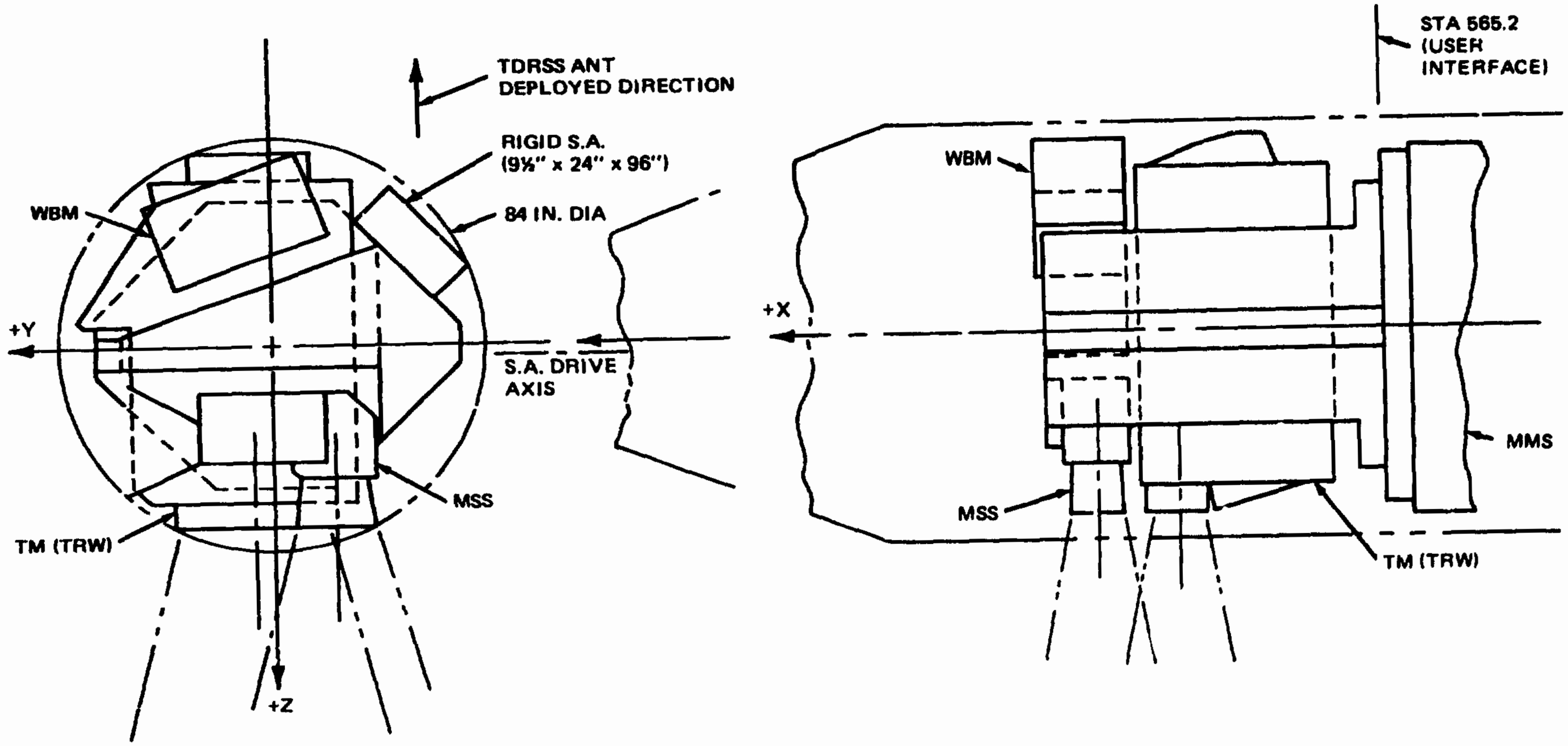

FIGURE 2.4-7 CONFIGURATION T-1B

2473-24

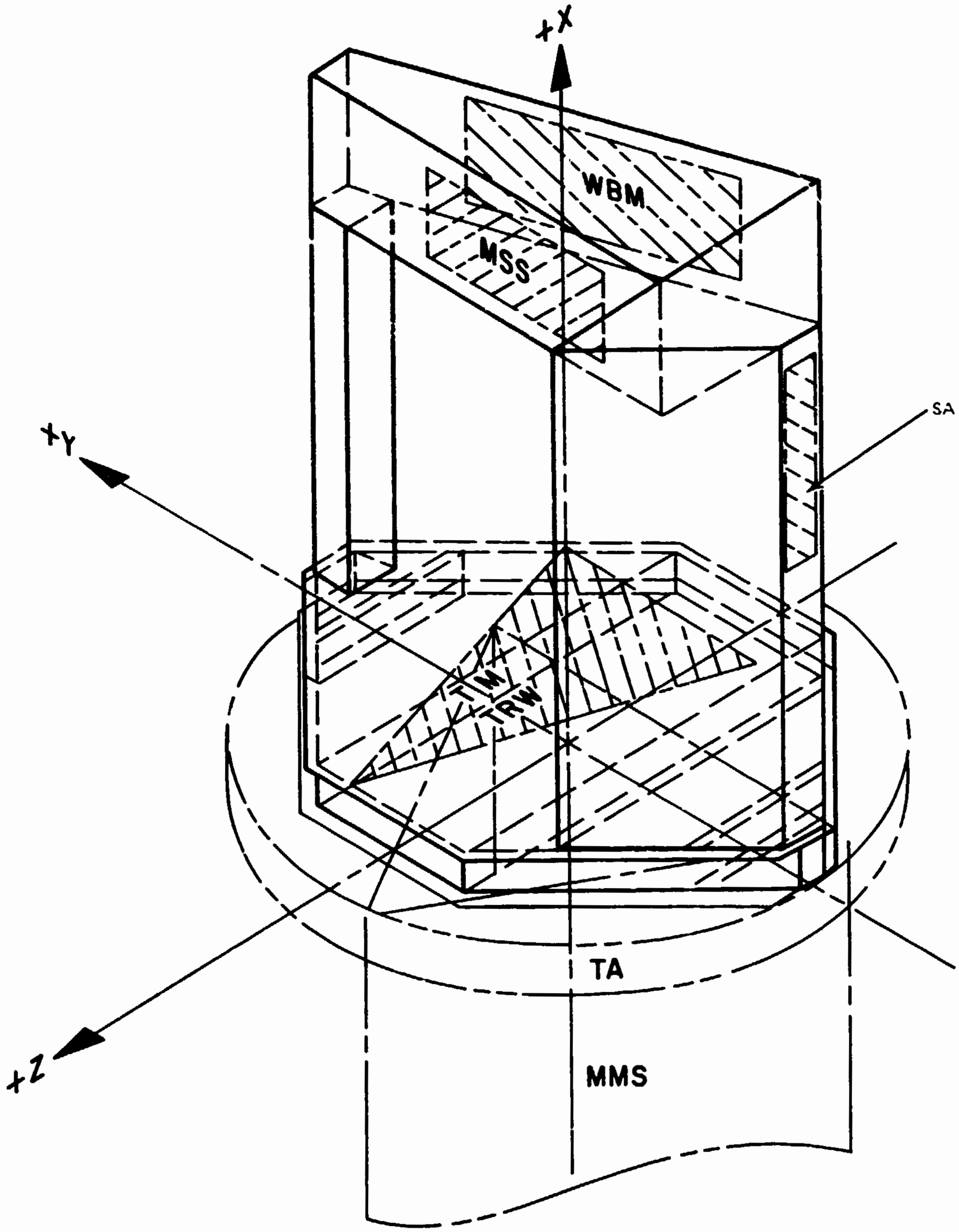

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FIGURE 2.4-8 STRUCTURE T-1B

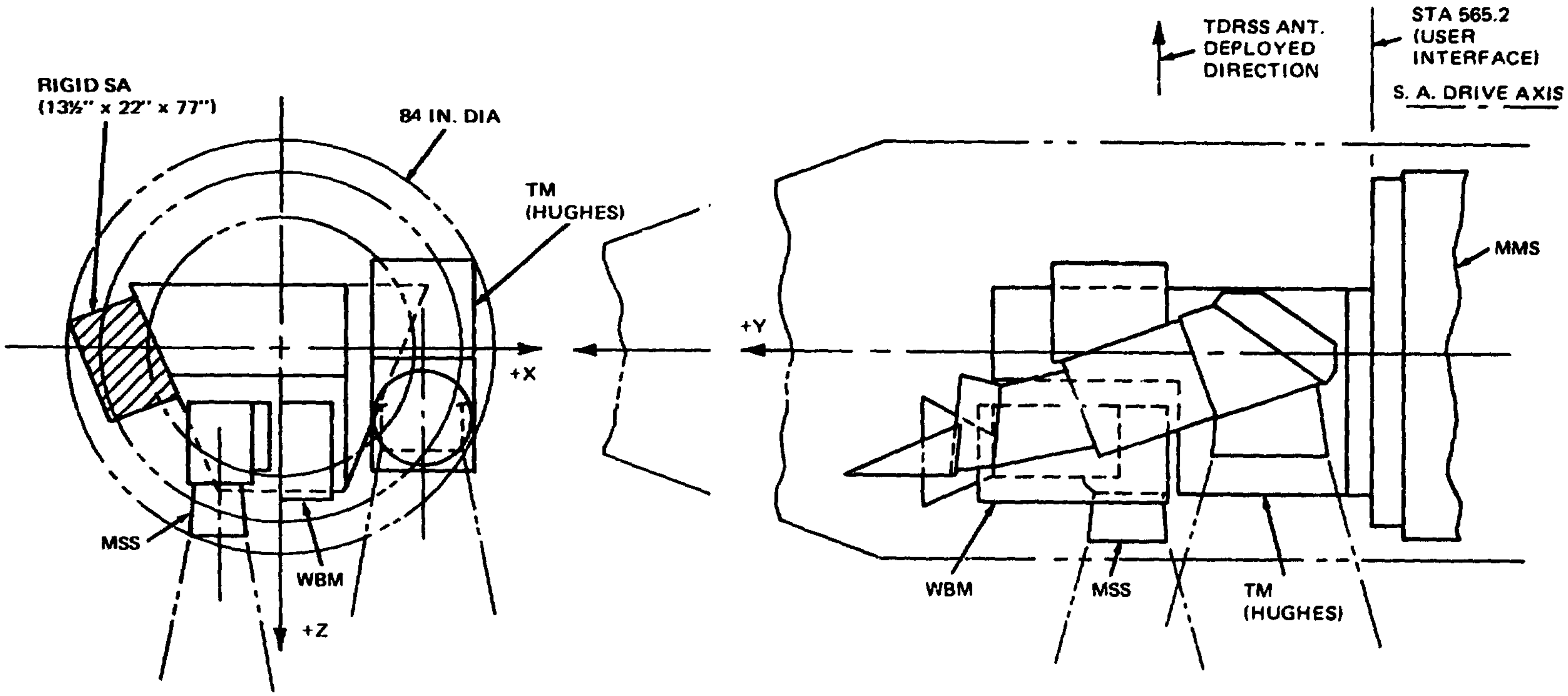

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FIGURE 2.4-9 CONFIGURATION H-2

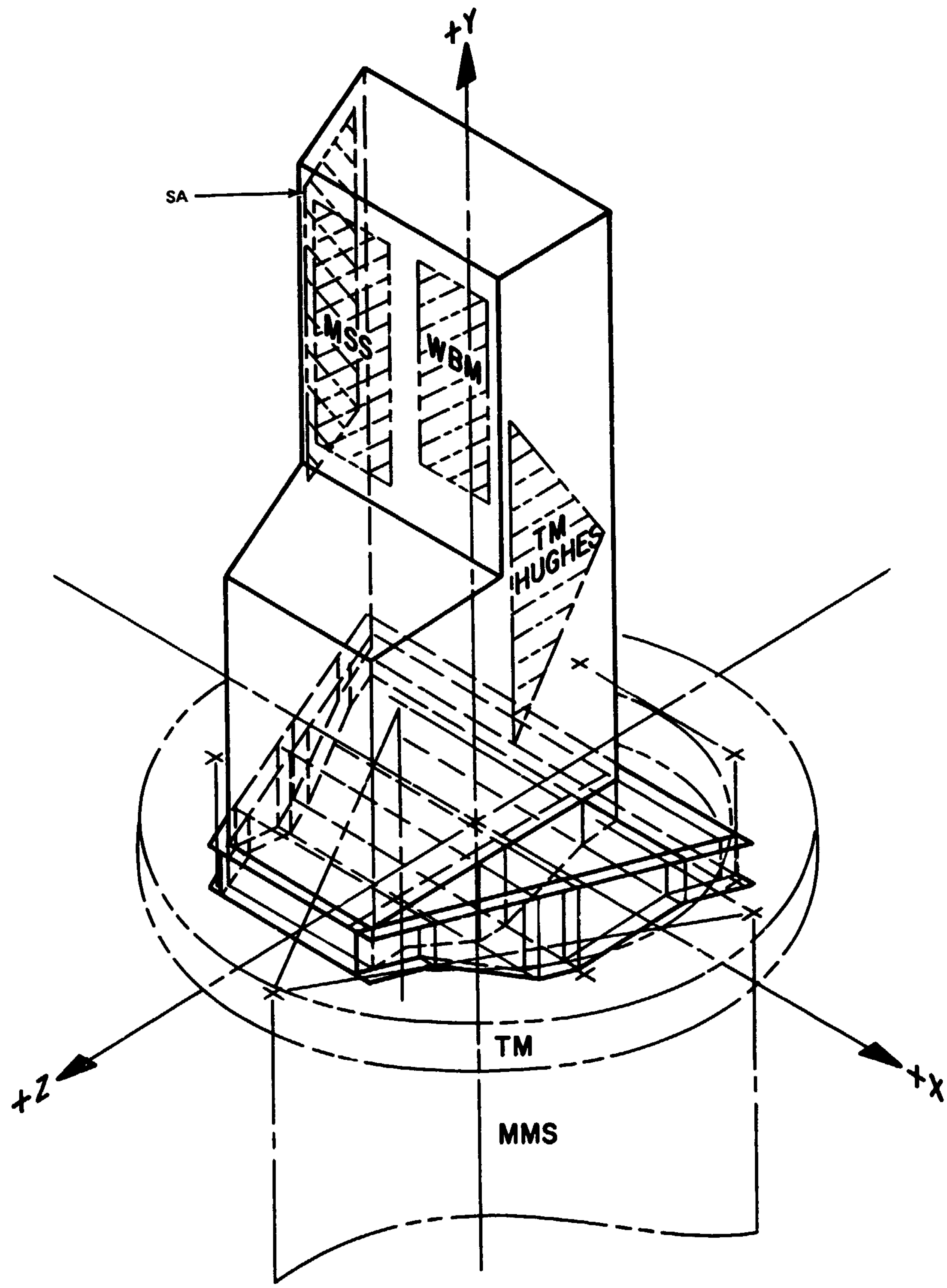

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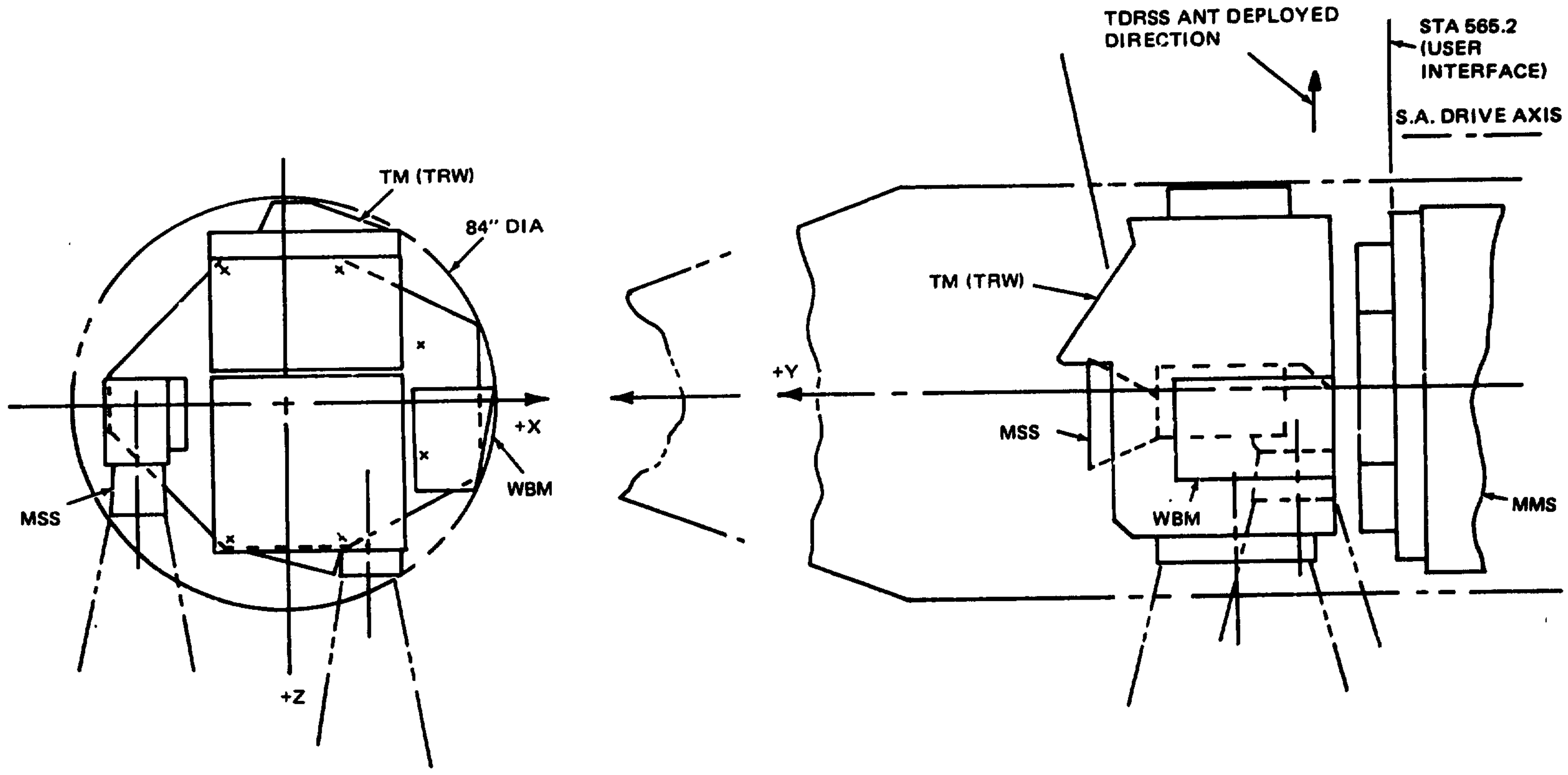
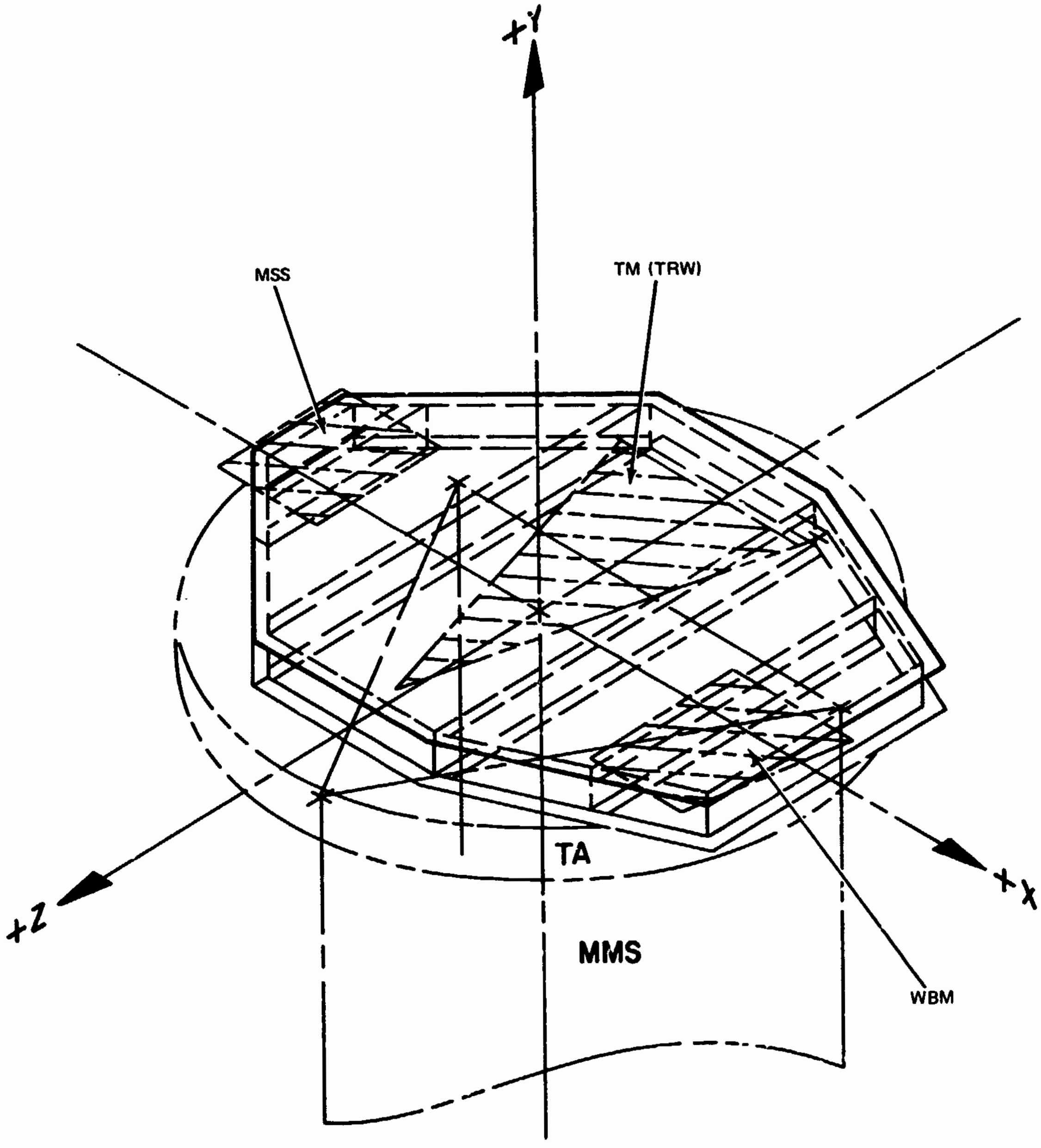
FIGURE 2.4-10 STRUCTURE H-2



2473-28

FIGURE 2.4-11 CONFIGURATION T-2



2473-29

FIGURE 2.4-12 STRUCTURE T-2

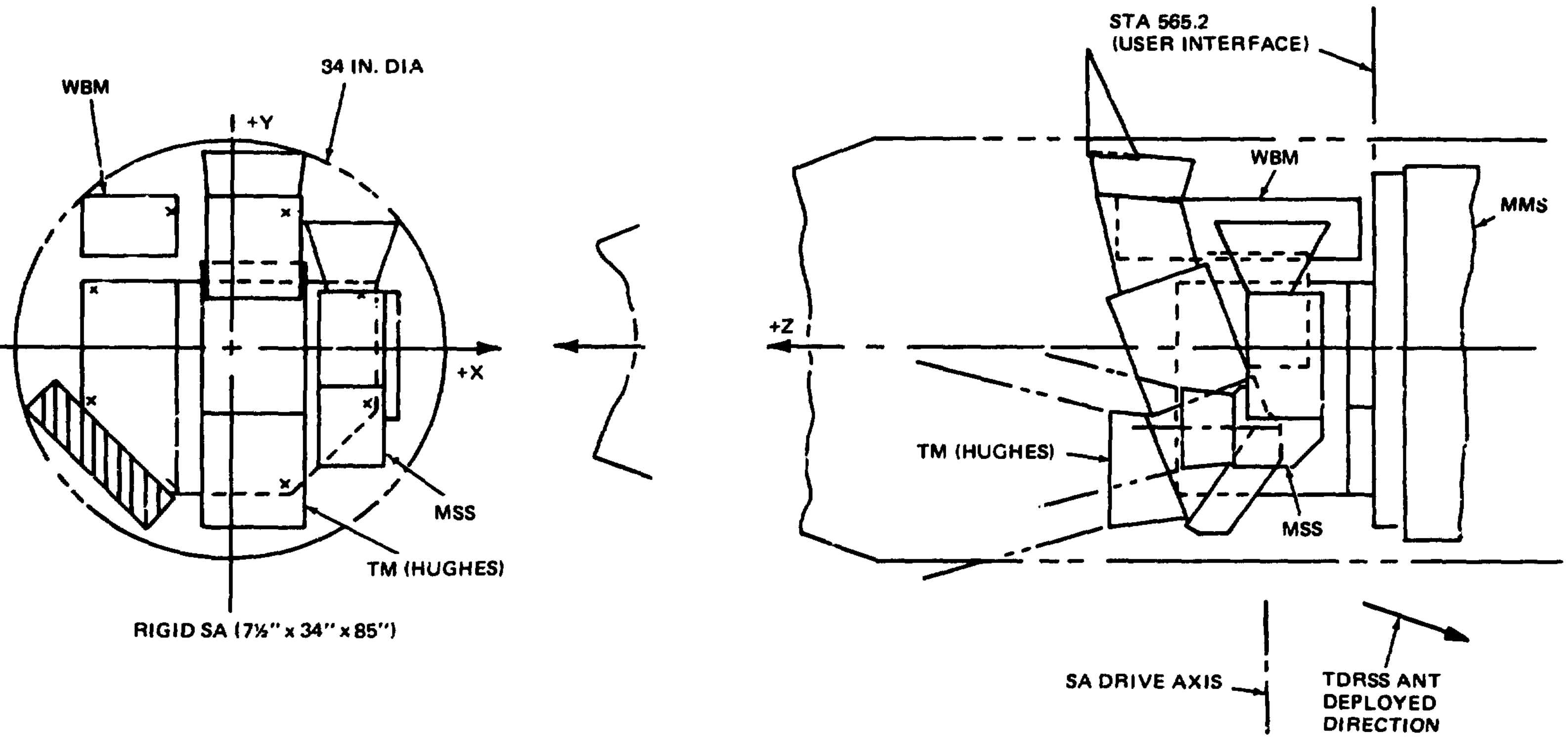


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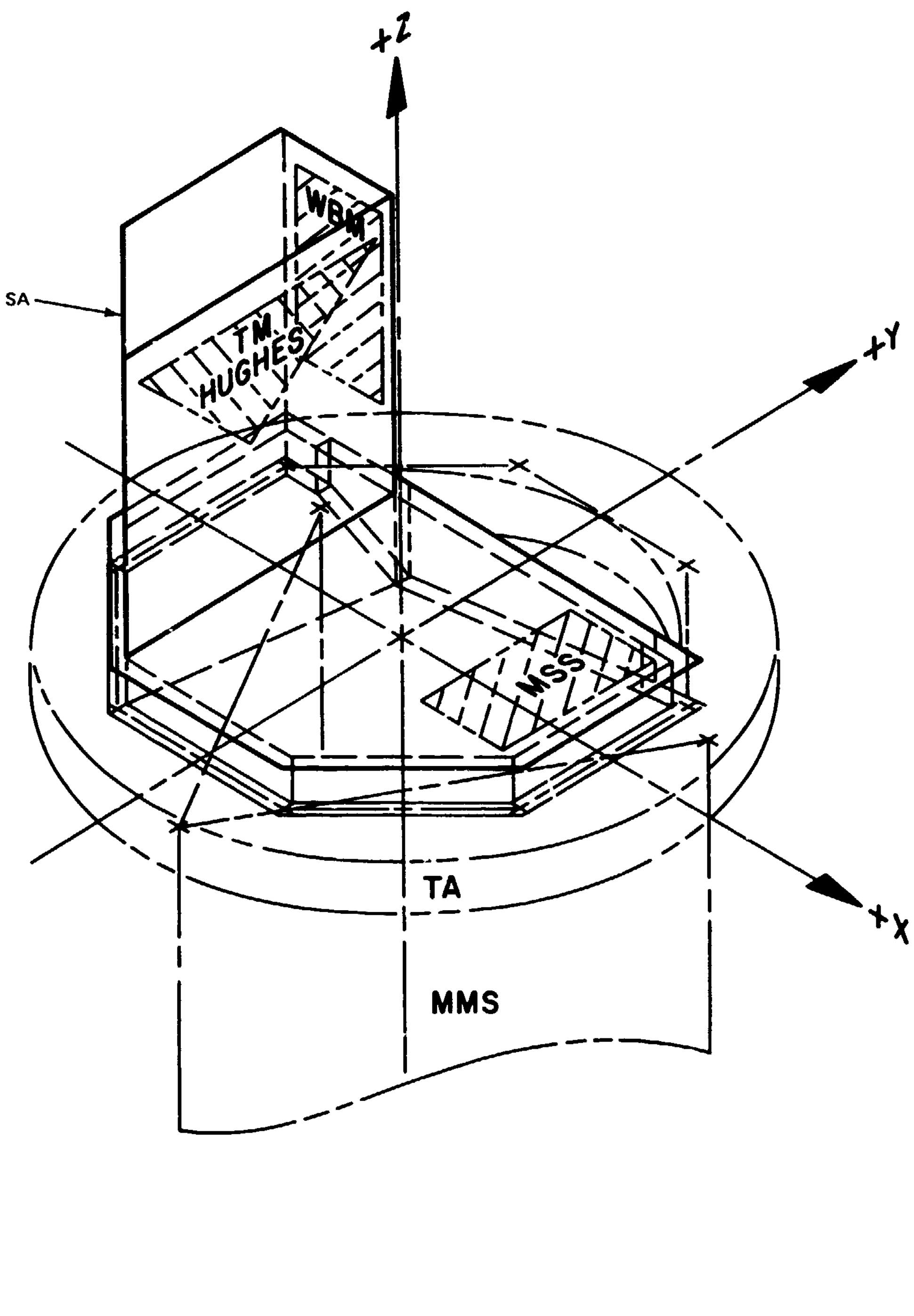
FIGURE 2.4-13 CONFIGURATION H-3



2473-31

FIGURE 2.4-14 STRUCTURE H-3

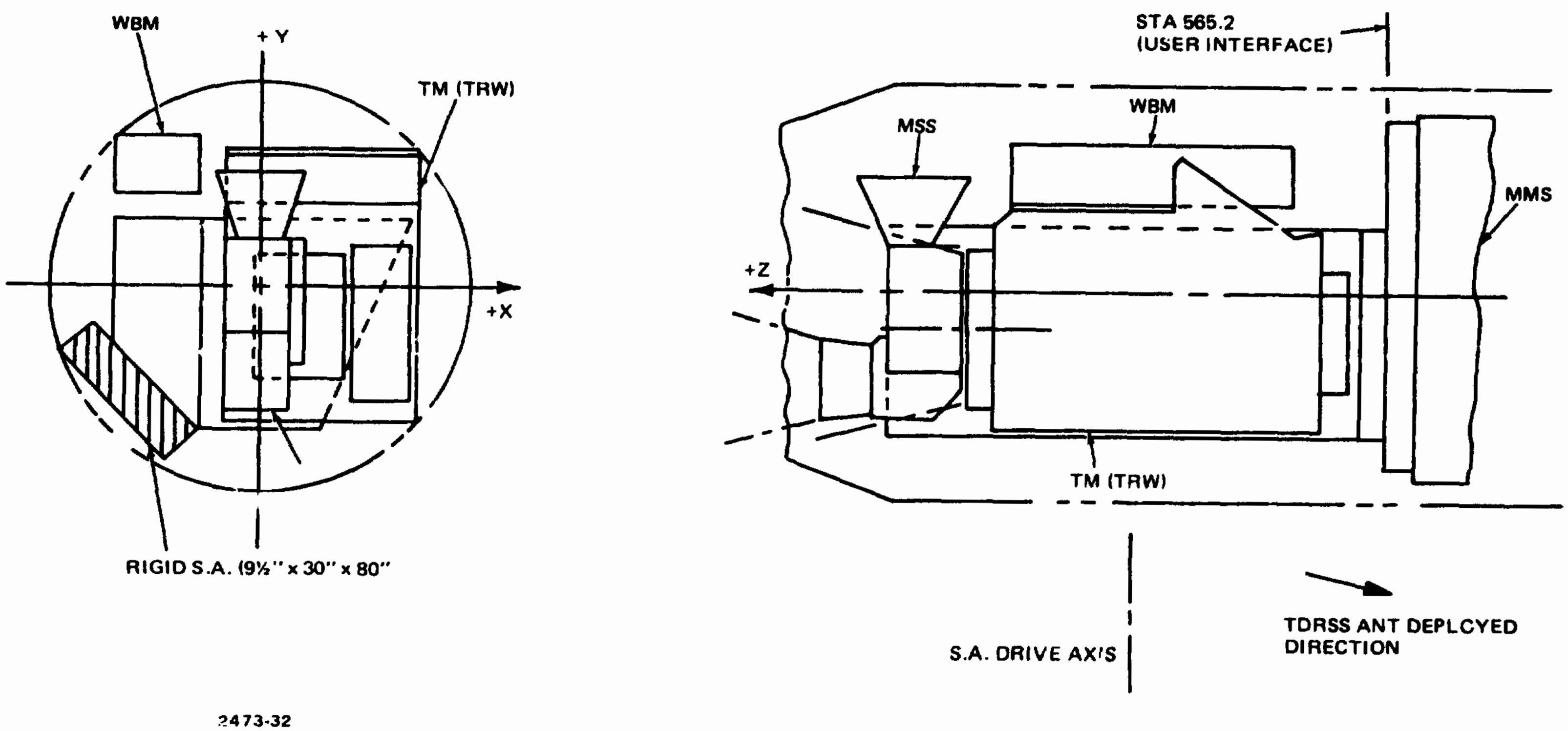
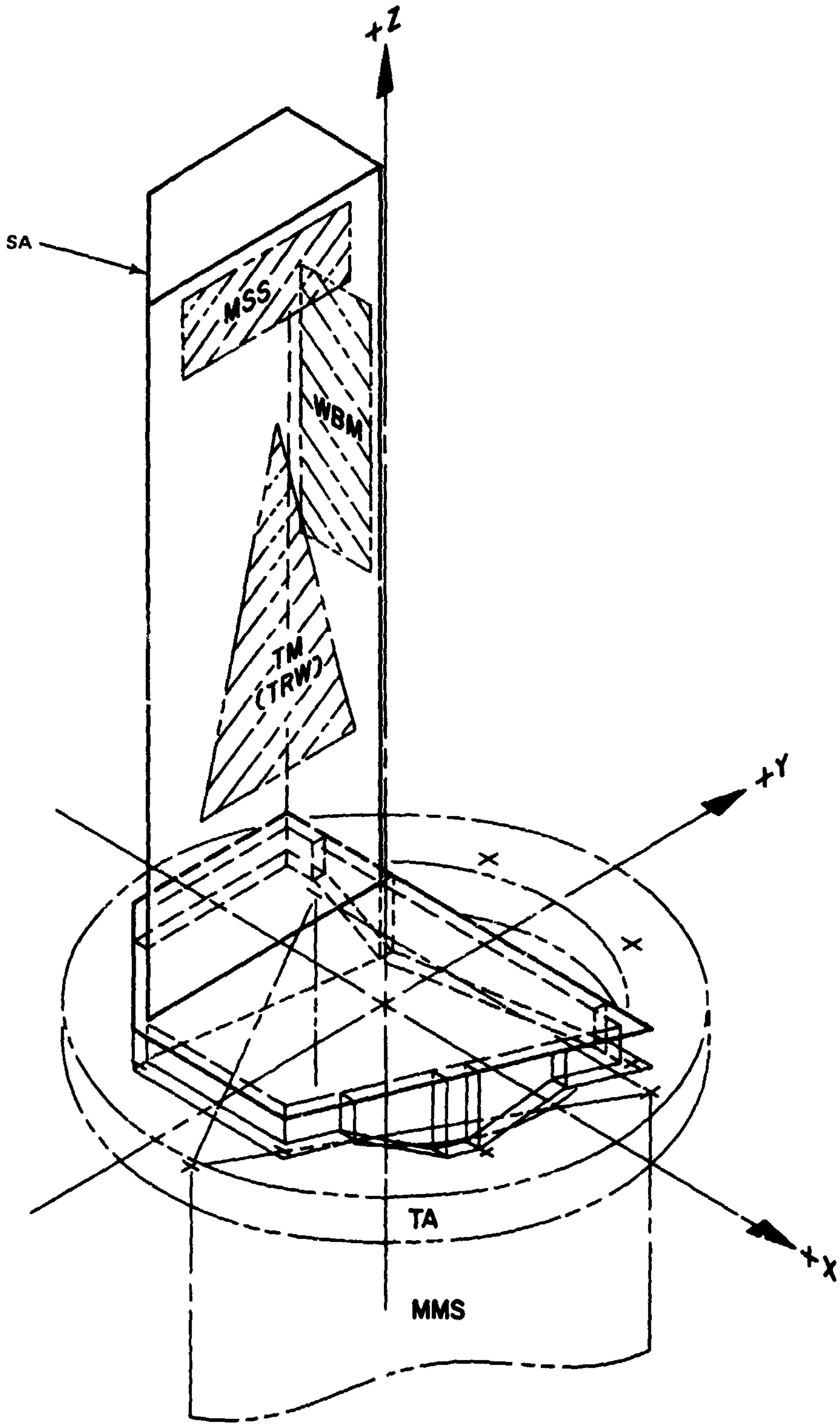


FIGURE 2.4-15 CONFIGURATION T-3



2473-33

FIGURE 2.4-16 STRUCTURE T-3

TABLE 2.4-1 INITIAL WEIGHT SUMMARY (POUNDS)

	HUGHES	TRW
MMS Spcft.	1306	1306
Propulsion Module	207	207
Batteries	150	150
Delta Vehicle Adapter	110	110
TDRS Antenna System	180	180
Solar Array	150	150
Wideband Module	110	110
Five Band MSS	148	148
Mechanism	100	100
Harness	75	75
Instrument Module Structure	230	230
Thermal Control System	75	75
Thematic Mapper	250	650
Contingency	579	179
TOTAL OBSERVATORY WEIGHT	3670	3670

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TABLE 2.4-2 ALTERNATE STOWED MASS PROPERTIES

		CG - in.			*	*	*	*
Configuration	Wt., Lb	X	Y	Z	Ixx	Iyy	Izz	Ixz
H-1A	1497	42.71	− 1.83	− 4.66	0.84	1.53	1.49	0.010
H-1A+Δcont	1897	40.03	− 1.45	− 3.68	1.02	1.89	1.84	−0.009
T-1A	1897	50.27	− 1.99	− 2.76	1.06	2.15	2.37	0.044
H-1B	1497	40.05	− 4.35	− 1.97	0.79	1.43	1.53	−0.052
H-1B+Δcont	1897	37.93	− 3.43	− 1.56	0.96	1.76	1.87	−0.058
T-1B	1897	44.59	− 2.45	1.39	1.01	2.34	2.55	0.069
H-2	1497	0.75	43.85	3.56	1.37	0.87	1.74	0.022
H-2+Δcont	1897	0.59	40.93	2.81	1.73	1.04	2.10	0.023
T-2	1897	3.00	35.21	− 2.08	1.32	1.14	1.74	−0.047
H-3	1497	− 1.43	− 1.53	37.02	1.36	1.52	0.85	−0.094
H-3+Δcont	1897	− 1.13	− 1.21	35.54	1.67	1.83	1.02	−0.097
T-3	1897	0.81	− 2.36	50.44	2.52	2.68	0.93	−0.042

*All inertias are lb(m)-in^{2}x10^{-6}

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TABLE 2.4-3 ALTERNATE DEPLOYED MASS PROPERTIES

Configuration	Wt., Lb	CG—in X	CG—in Y	CG—in Z	* Ixx	* Iyy	* Izz	* Ixz	S.A. Position	Ant. Mast Length, Ft
H-1A	1497	37.53	−28.74	−23.25	9.37	4.20	6.96	0.100	A	10
H-1A+Δcont	1897	35.94	−22.68	−18.34	9.96	4.68	7.54	0.045	A	10
T-1A	1897	41.03	−18.84	−17.33	9.55	4.22	7.97	0.621	A	10
H-1A	1497	37.53	−28.74	− 2.80	10.16	5.00	6.96	0.543	B	10
H-1A+Δcont	1897	35.94	−22.68	− 2.21	10.59	5.31	7.54	0.537	B	10
T-1A	1897	41.03	−18.84	− 1.60	11.75	6.43	7.97	−0.036	B	10
H-1A	1497	42.74	−28.74	−13.02	8.36	3.78	7.56	0.423	C	10
H-1A+Δcont	1897	40.05	−22.68	−10.28	8.84	4.19	8.16	0.371	C	10
T-1A	1897	45.14	−18.84	− 9.66	9.26	3.96	8.00	0.571	C	10
H-1A	1497	27.31	−28.74	−13.02	8.36	4.15	7.93	0.122	D	10
H-1A+Δcont	1897	27.87	−22.68	−10.28	8.84	4.51	8.49	0.134	D	10
T-1A	1897	32.97	−18.84	− 9.66	9.26	6.04	10.08	−0.230	D	10
H-1A	1497	37.53	−28.74	−33.43	14.32	9.16	6.96	0.398	A	20
H-1A+Δcont	1897	35.94	−22.68	−26.38	15.10	9.82	7.54	0.318	A	20
T-1A	1897	41.03	−18.84	−25.37	14.72	9.39	7.97	0.972	A	20

*All inertias are $lb(m) \cdot in^2 x 10^{-6}$

2473-36T

SOLAR ARRAY POSITIONS

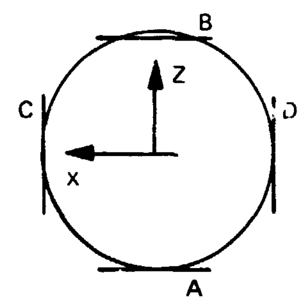

TABLE 2.4-4 WORST CASE MASS PROPERTIES WITH MMS

Solar Array in Position 'A'

Configuration	Wt (lb)	CG—in X	CG—in Y	CG—in Z	* Ixx	* Iyy	* Izz	* Ixz	Ant. Mast Length, Ft
H-1A	3270	4.17	−13.16	−10.69	11.16	8.48	11.48	−1.061	10
H-1A+Δcont	3670	6.98	−11.72	− 9.48	11.42	9.06	12.07	− .963	10
T-1A	3670	9.61	9.74	− 8.96	10.83	9.14	12.94	− .412	10
H-1A	3270	4.17	−13.16	−15.30	16.58	13.91	11.48	−1.271	20
H-1A+Δcont	3670	6.98	−11.72	−13.63	16.89	14.53	12.07	−1.130	20
T-1A	3670	9.61	− 9.74	−13.11	16.31	14.63	12.94	− .540	20

*All inertias ar $lb(m) \cdot in^2 x 10^{-6}$

2473-37T

This major shift in the cg is due to the unbalanced but necessary positioning of the appendages in their orbital positions. The orbital adjust and attitude control system must be capable of adjusting the thrust vector for this geometry shift.

2.5 ORBIT APPENDAGE RELATIONSHIP ANALYSES

In parallel with the configuration study during Phase I, an investigation was conducted to determine the orbital positional inter-relationships between the solar array, the TDRSS antenna, the earth, the sun, the TDRSS satellite, the Landsat spacecraft and the experiment fields-of-view.

Initially it appeared that the appendage requirements would be harsh. In order to obtain a full 232° FOV (complete sky coverage minus earth occultation), the TDRSS antenna would require a mast height in excess of 20 feet. This would require an extremely complex deployment mechanism arrangement coupled with unfavorable and unbalanced mass properties. In addition, in order to minimize the antenna height requirement, a trapezoidal rigid array appeared necessary. A stowed volumetric analysis indicated that the rigid array concept, though unwieldy, was feasible. Figure 2.5-1 shows the volumetric analysis results. In an attempt to simplify the appendage designs, a more comprehensive search was made into the orbital inter-relationships affecting these designs.

These analyses were conducted for the 9:30 am (study baseline) orbit to determine the constraints on the size and shape of the deployed solar array due to the TDRSS antenna requirements. They indicated that a rectangular array could be used with a properly designed mechanical drive and deployment system. The impact of this investigation was significant in the design and ultimate reduction of cost to the Landsat Program. First, the acceptability of a rectangular array allows a modified off-the-shelf FRUSA array system to be used (as opposed to a custom-made, rigid shaped array). Second, the TDRSS antenna height was reduced to approximately 50-60% of the height originally thought necessary.

This reduced antenna height allowed a simplified antenna mast deployment system to be developed in Phase II of this program. A comprehensive investigation was performed to determine the interrelationship between the orbit, the appendages, and the earth. Figures 2.5-2a through 2.5-2c illustrate some of the geometric relationships investigated.

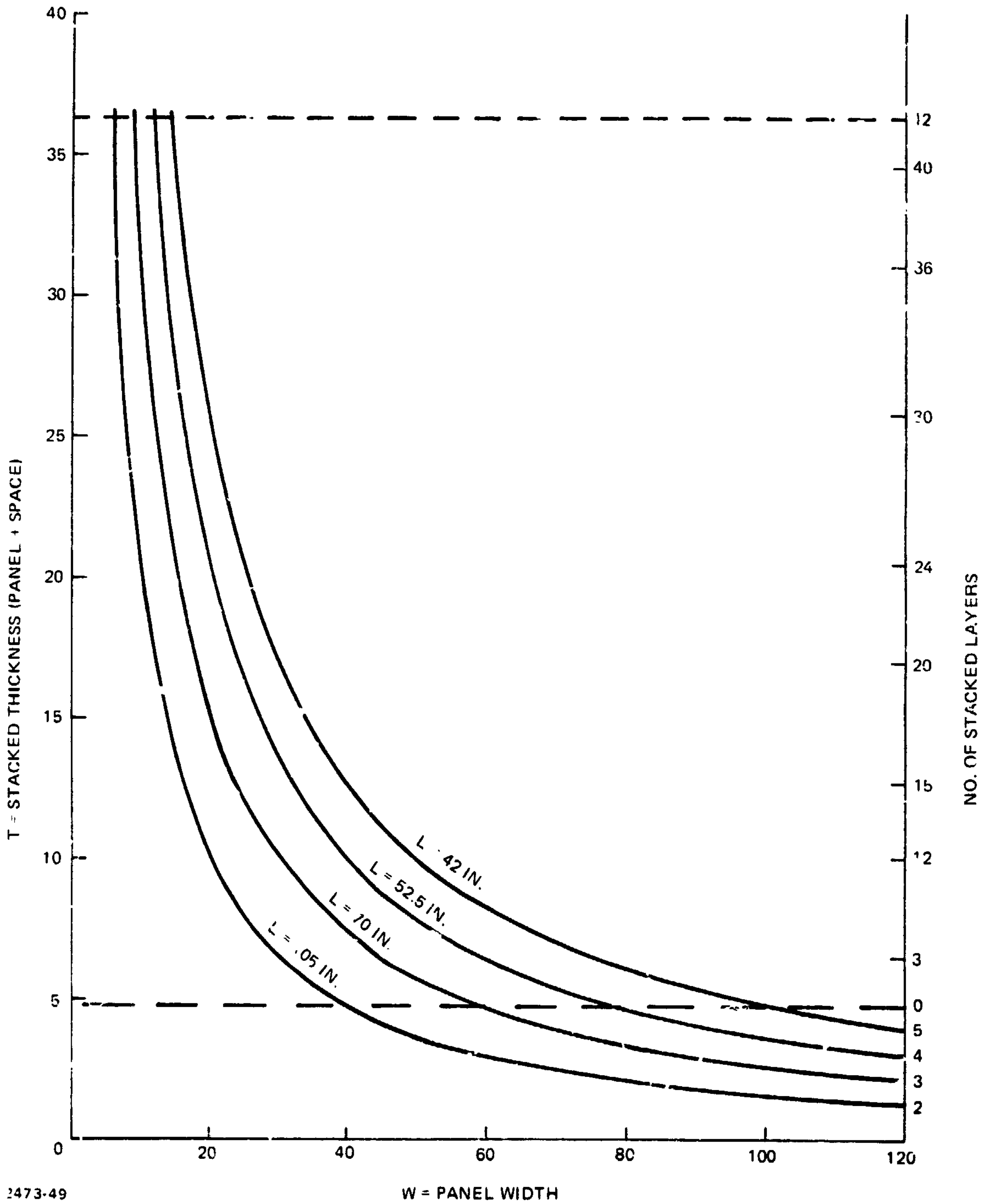

FIGURE 2.5-1 RIGID SOLAR ARRAY STOWED VOLUMETRICS

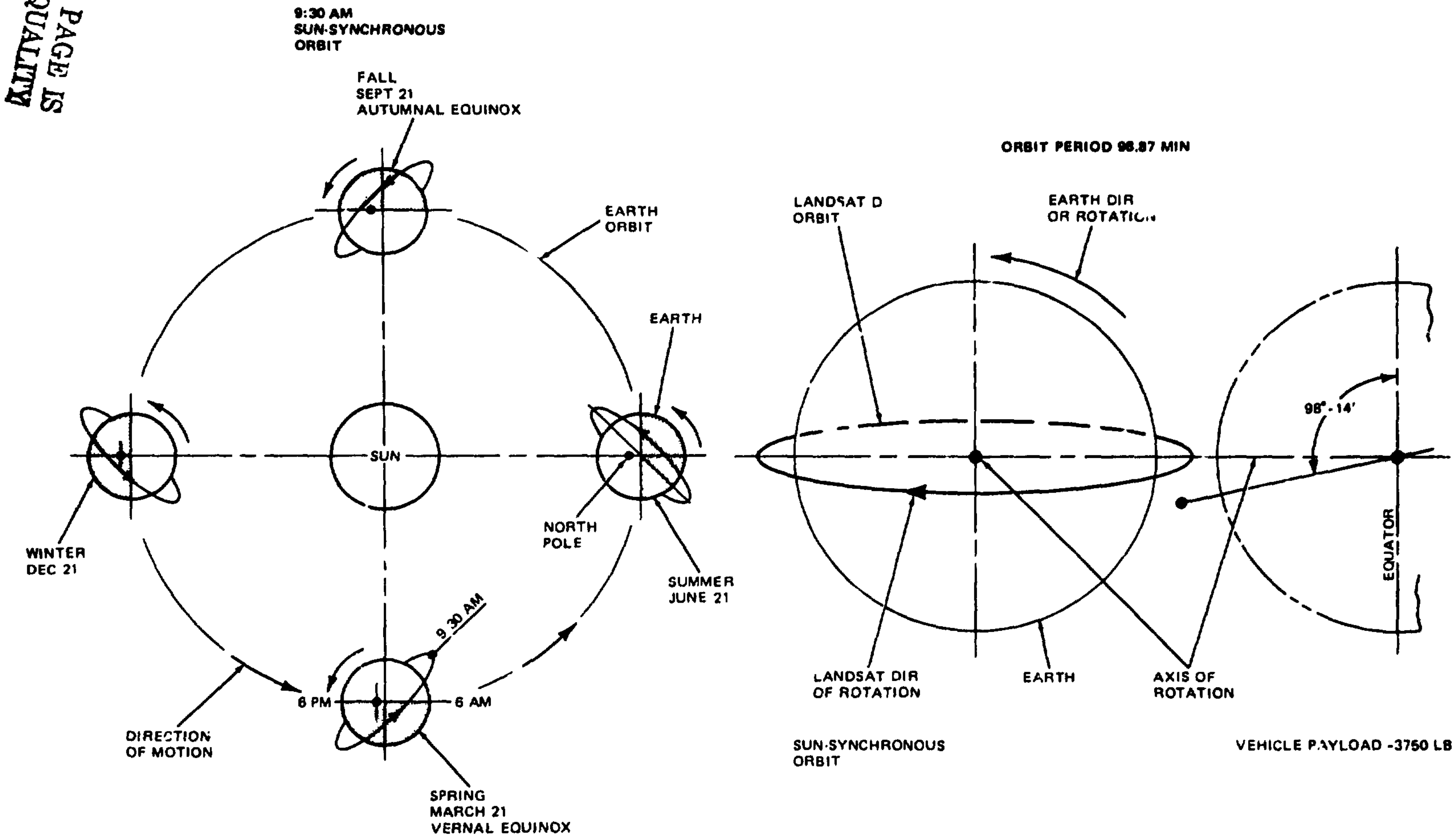

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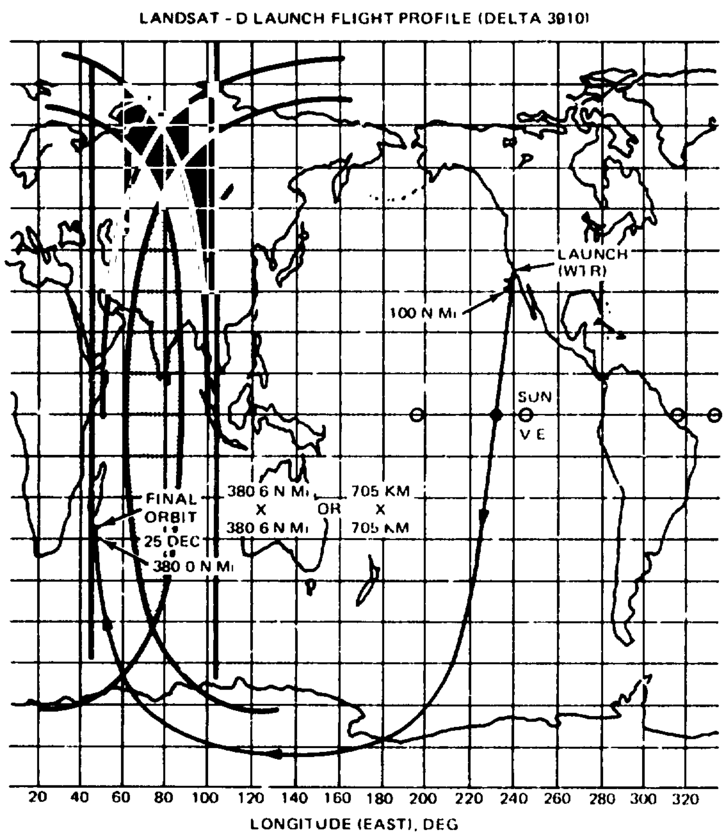
FIGURE 2.5-2A ORBITAL RELATIONSHIPS



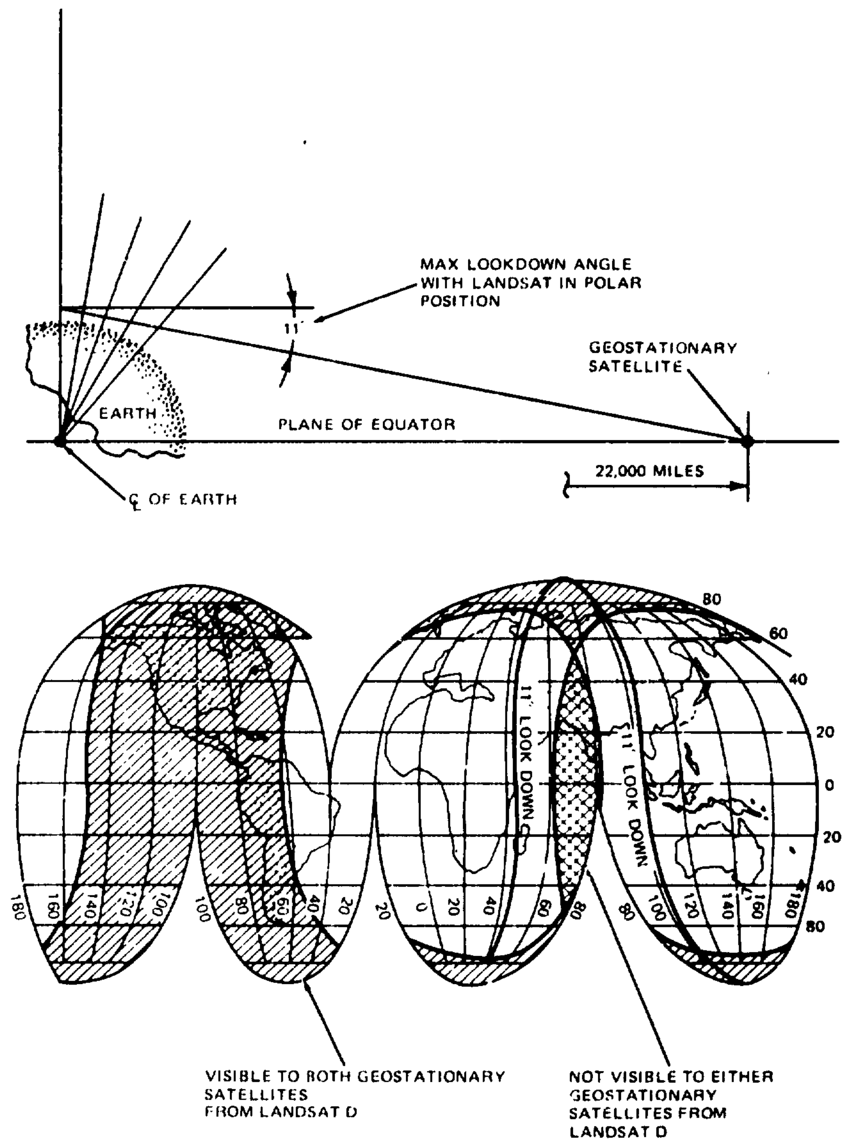


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FIGURE 2.5-22 ORBITAL RELATIONSHIPS

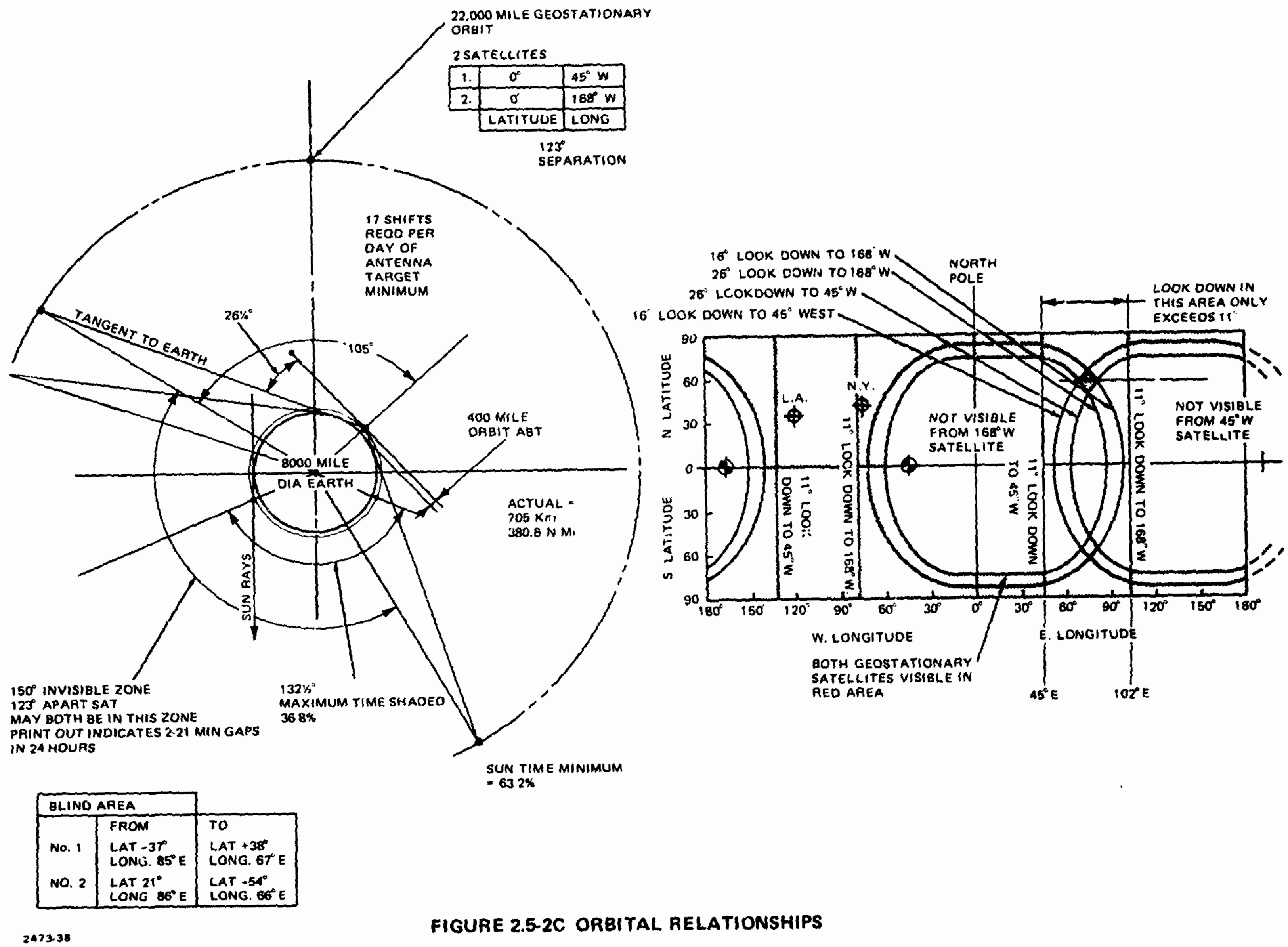


FIGURE 2.5-2C ORBITAL RELATIONSHIPS

Figure 2.5-3 displays the specific results of this in-depth study. Listed are the specific appendage relationships as a function of the earth's latitude.

2.6 APPENDAGE OPTIONS

As a direct result of the orbital study, simplified concepts were developed for the deployment of the Solar Array (FRUSA) and the TDRSS antenna mast. Figures 2.6-1a through 2.6-1g indicate the alternate geometry interplay between the antenna, the array and the experiment fields-of-view. Figures 2.6-2a and 2.6-2b indicate two of the various array options considered with respect to stowage and orbital orientation. Figures 2.6-3a through 2.6-3f show some of the mechanism options that were studied.

It can be seen that an offset crank was added to the array gimbal axis to minimize the array blockage of the antenna FOV.

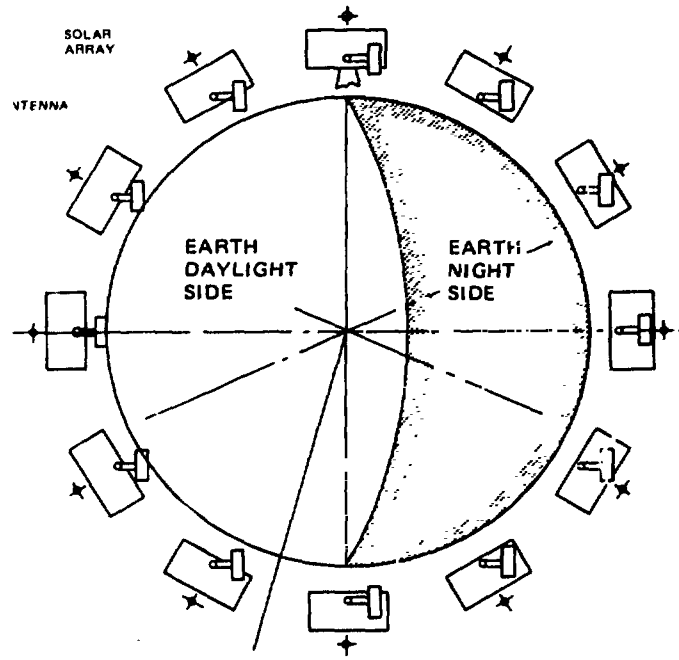
Using the deployed position required by orbit analysis and the stowed volume requirements of the shroud, a number of potential solar array configurations and associated deployment schemes were developed. Figures 2.6-4 through 2.6-7 represent these concepts. This conceptual development was taken only as far as was necessary to demonstrate design feasibility. More detailed analysis of the appendage deployment systems will be found in the mechanical system section of Phase II

2.7 EVALUATION AND SELECTION

After the eight configurations, which broadly satisfied the Landsat criteria were established, there remained the task of evaluation and selection of the most viable candidate for each of the two Thematic Mappers. Each of the configurations were reviewed and evaluated with respect to their merits for: configuration efficiency, structural capability, mechanical complexity, thermal design and a general potential for best satisfying mission requirements. Some of the specific key comparative evaluators are listed below.

- Structural efficiency
- Weight and Center-of-Gravity
- Accessibility for Refurbishment
- Commonality and Adaptability to Downstream equipment change
- Structural continuity and compatibility with MMS
- Equipment and Thematic Mapper mounting provisions

- IN DAYLIGHT ARRAY IS IN LOWER 180°
- FOR LATITUDE BELOW 50°N OR S, ARRAY BELOW CRIT ANT ANGLES
- FOR LATITUDE ABOVE 75°N OR S, ANT-LOOKDOWN ANGLE MOSTLY LESS THAN MAX, & CLEARS ARRAY
- FOR LATITUDE BETWEEN 50° & 75°N OR S, LOOKDOWN ANGLE MOSTLY LESS THAN 11° & NO PROB
- BETWEEN 50° & 75° S WHEN LOOKDOWN ANGLE LARGE–NO LAND MASSES OCCUR
- BET 50° & 75°N WHEN LOOKDOWN ANGLE LARGE, EITHER:
  (1)–ARRAY NOT IN CRIT ZONE
  (2)–EARTH OCCULTS BEFORE ARRAY



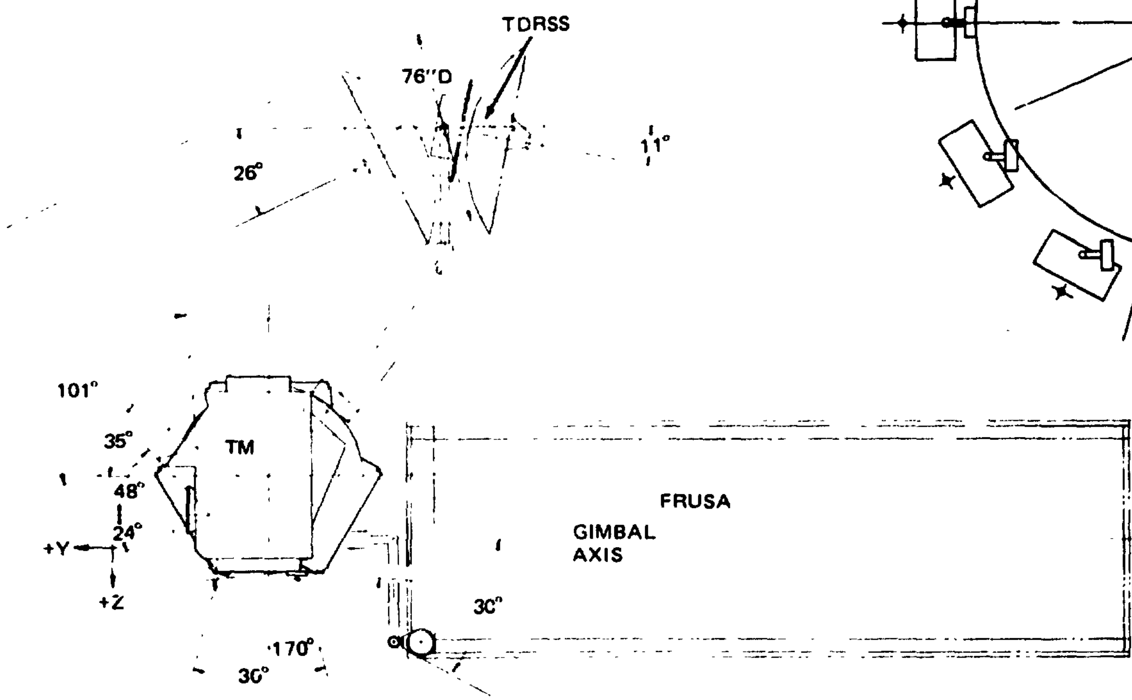


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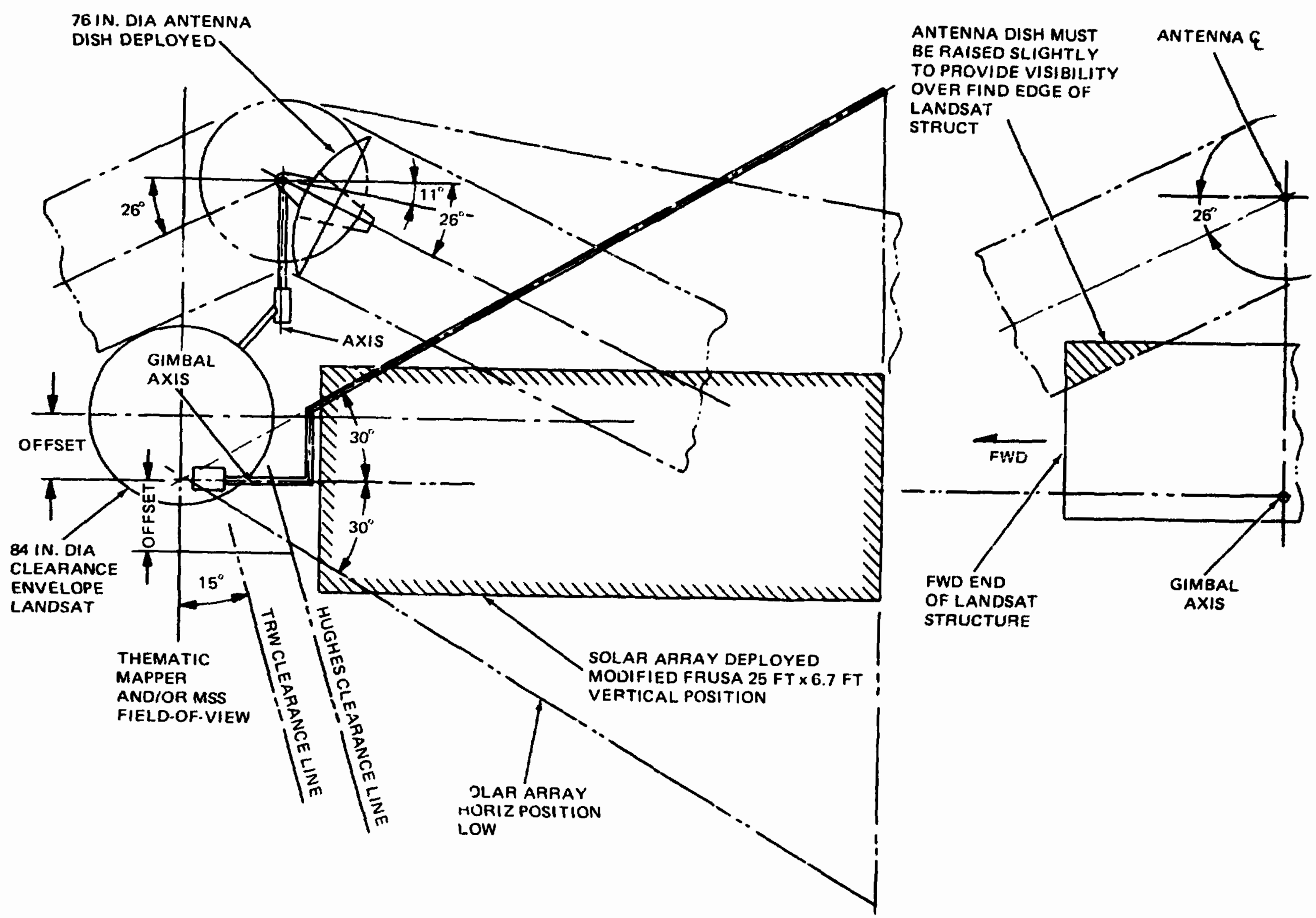
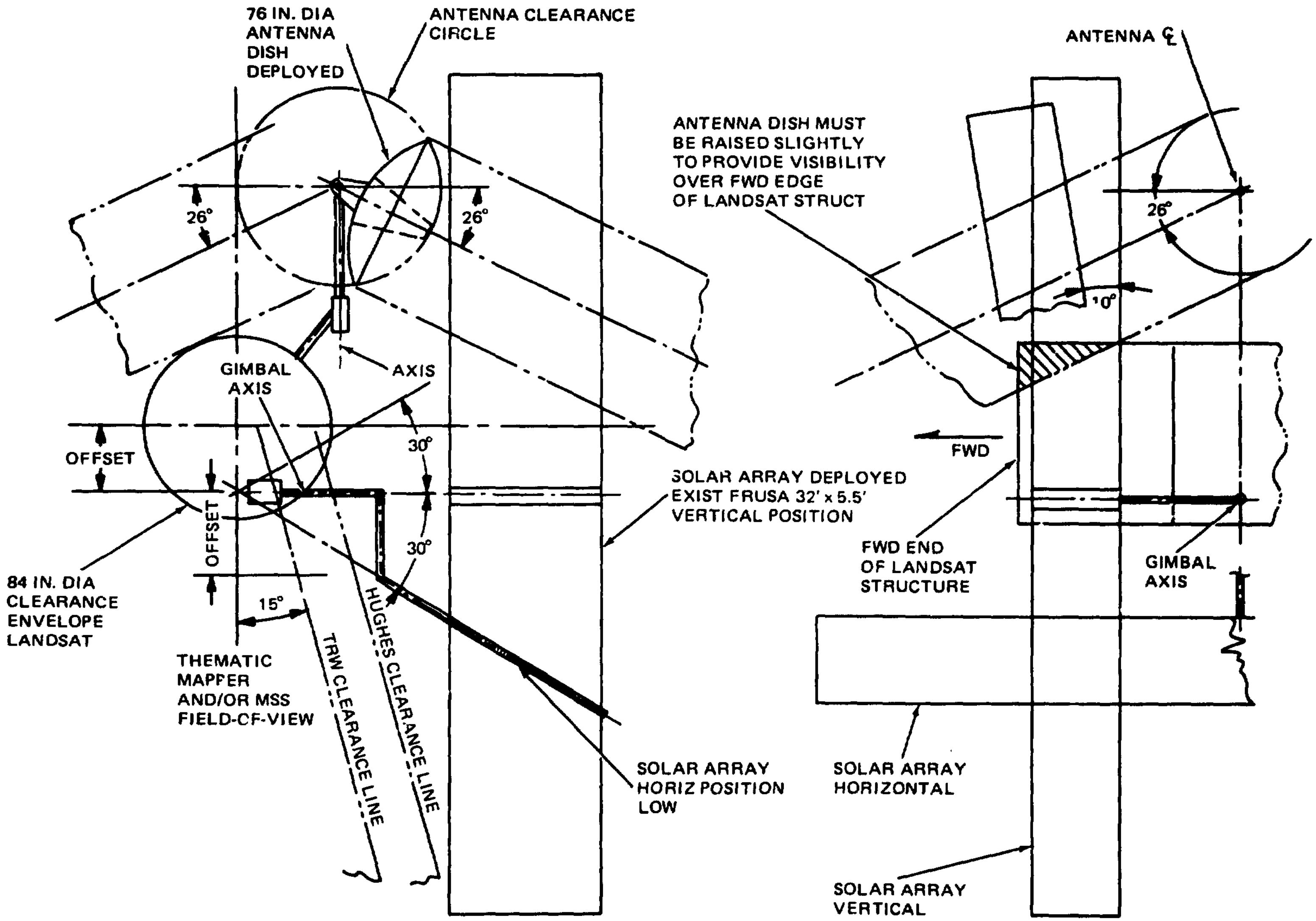
FIGURE 2.5-3 SOLAR ARRAY AND ANTENNA ORBITAL RELATIONSHIP



2473-44

FIGURE 2.6-1A APPENDAGE GEOMETRY



2473-45

FIGURE 2.6-1B APPENDAGE GEOMETRY

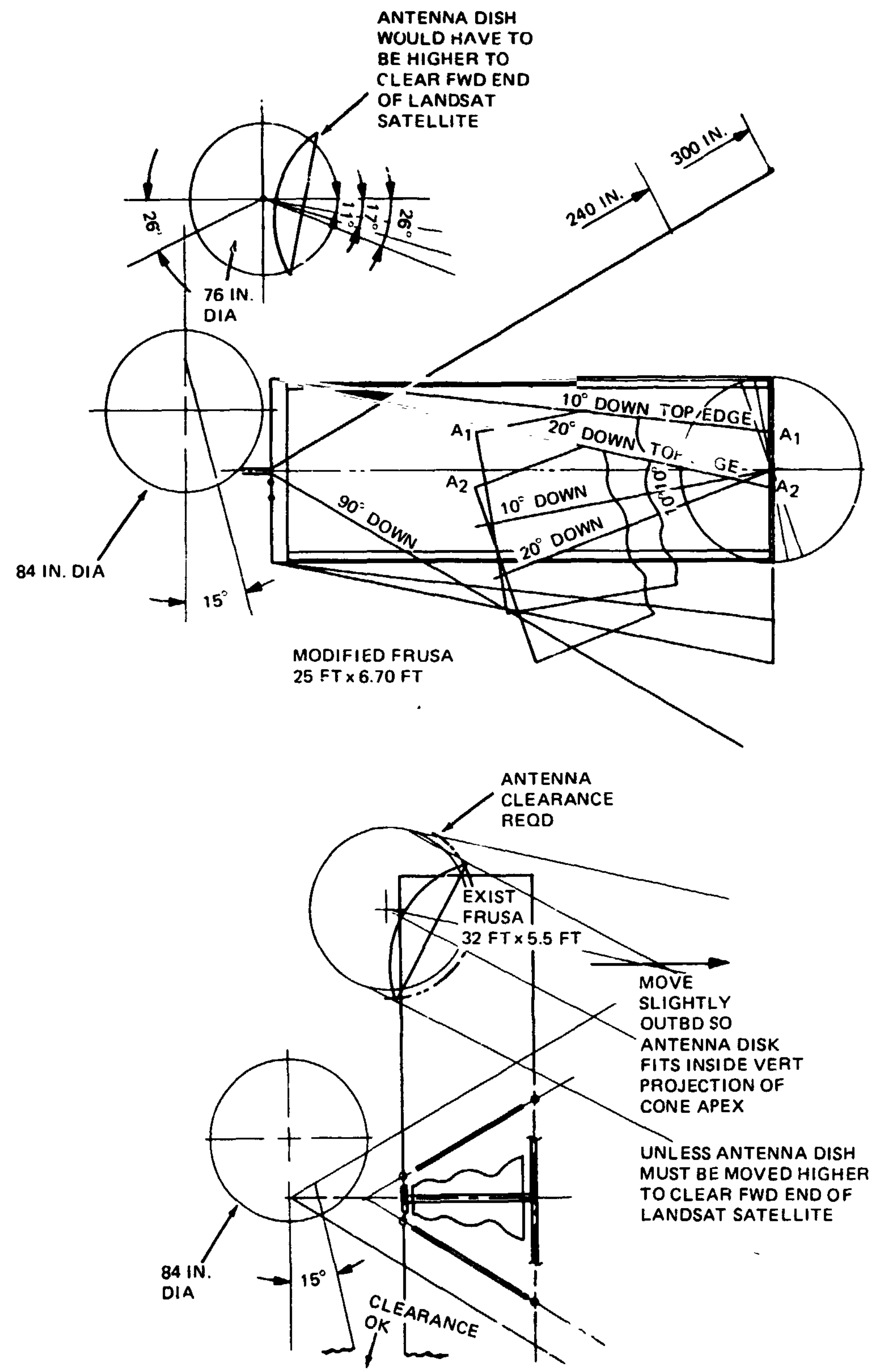


FIGURE 2.6-1C APPENDAGE GEOMETRY

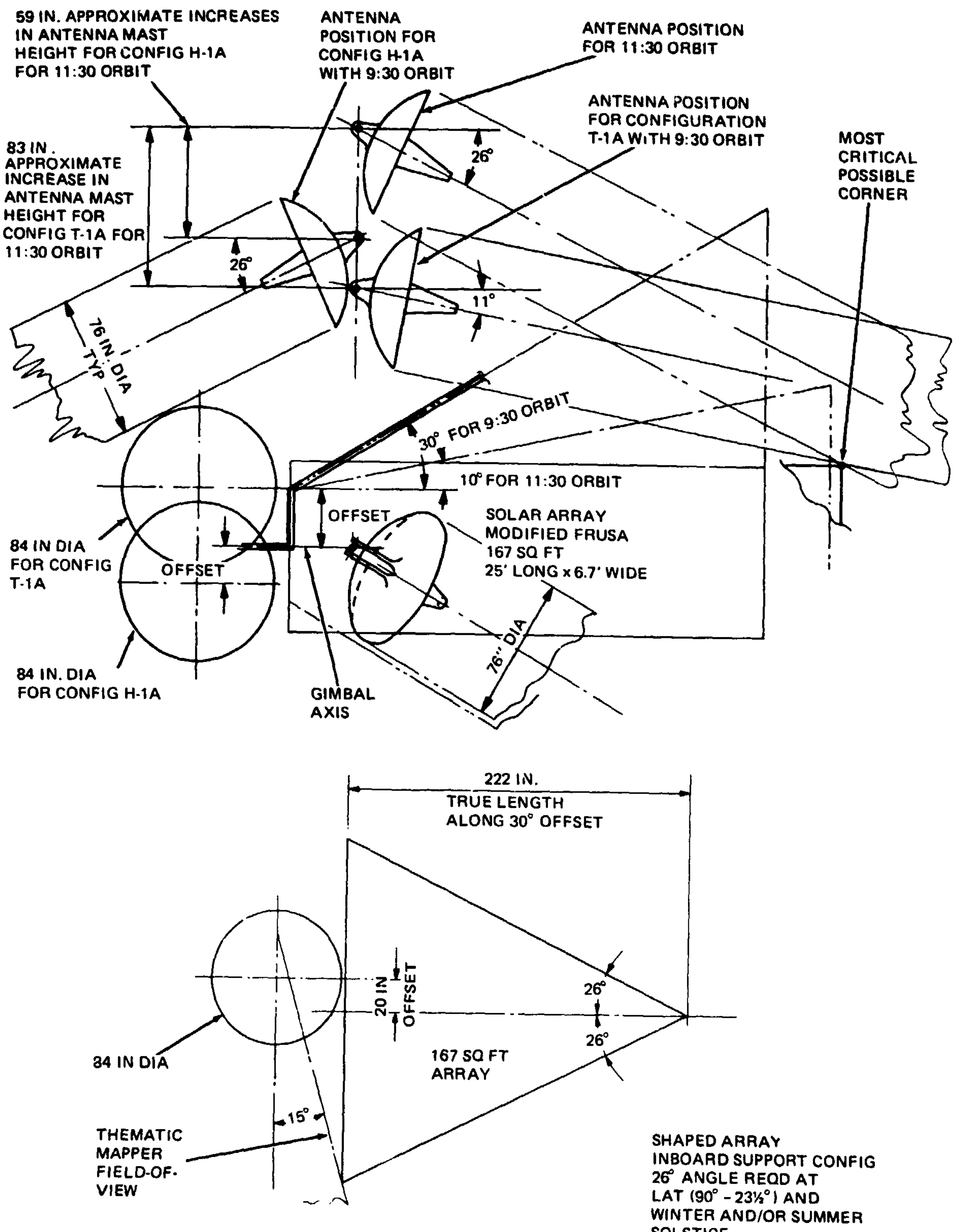


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FIGURE 2.6-1D APPENDAGE GEOMETRY

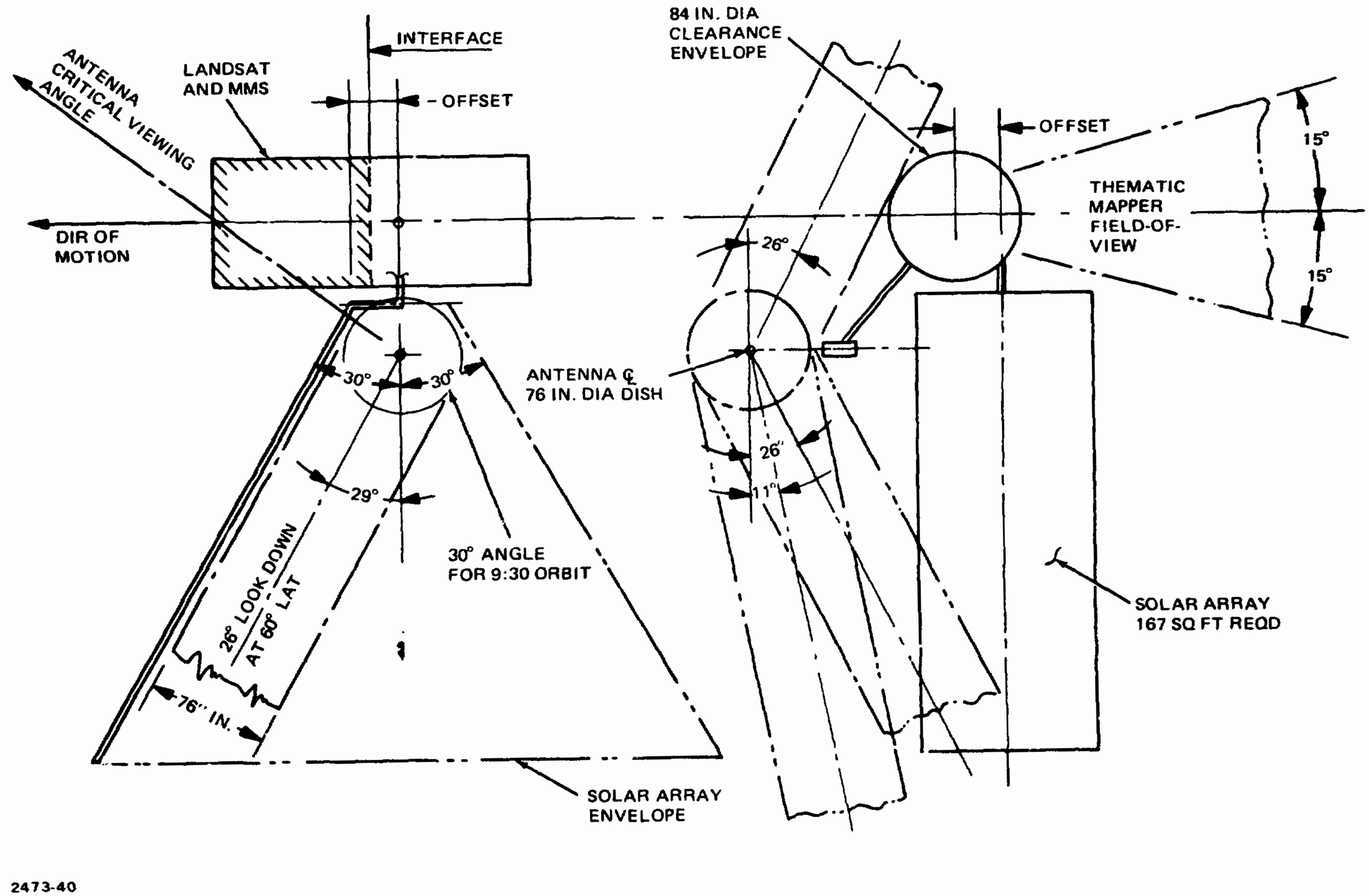

FIGURE 2.6-1E APPENDAGE GEOMETRY

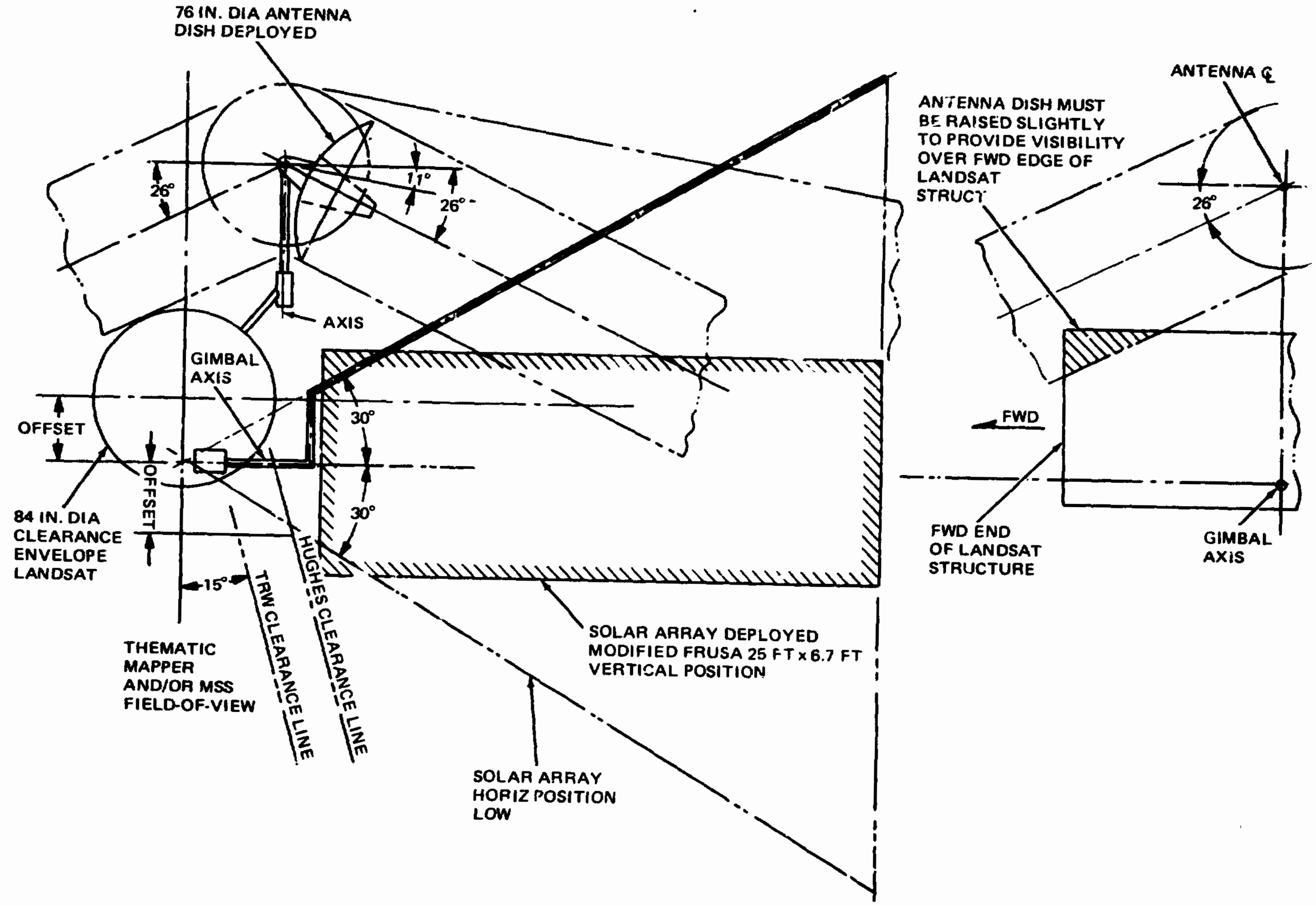

FIGURE 2.6-1F APPENDAGE GEOMETRY

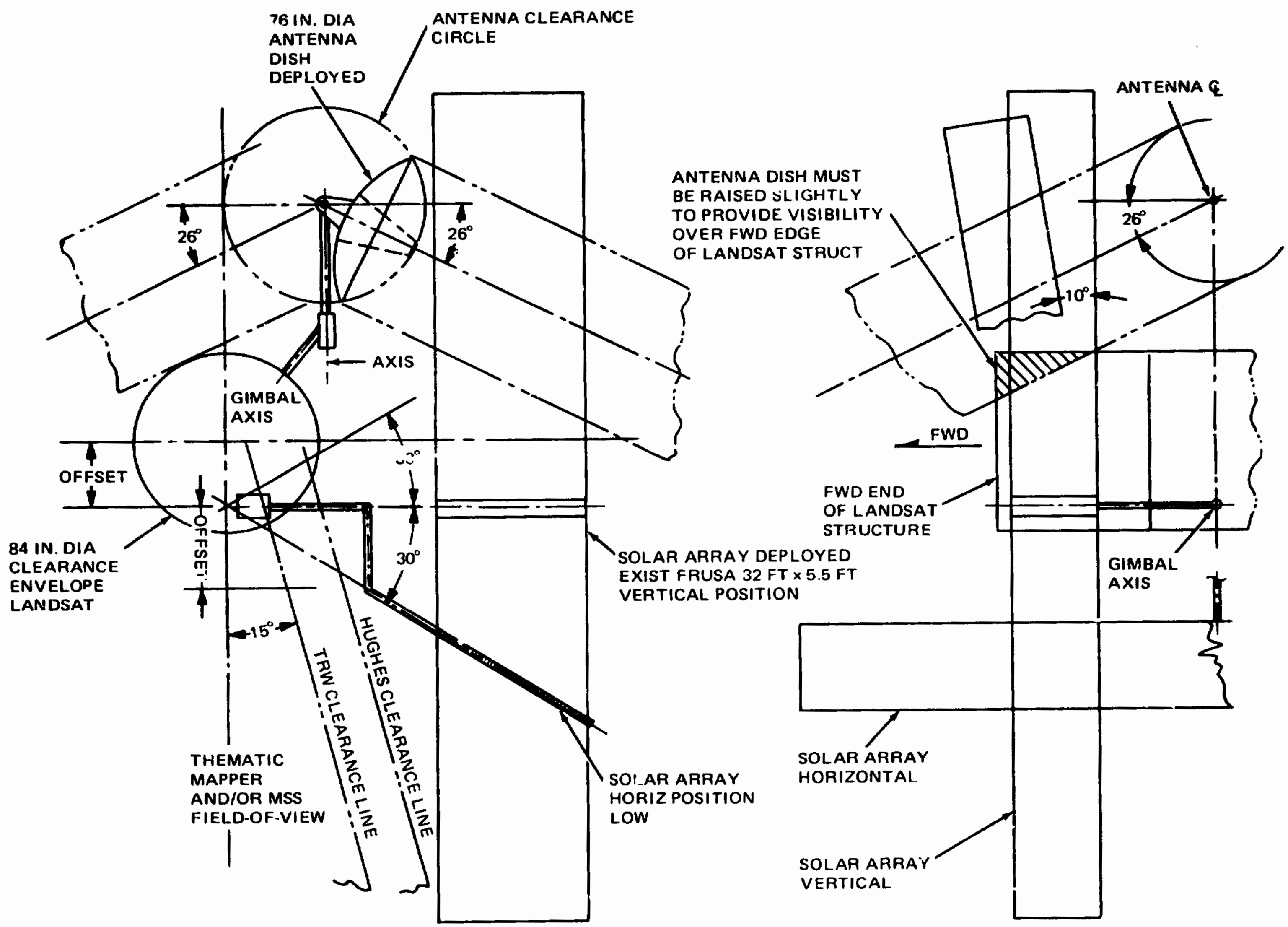

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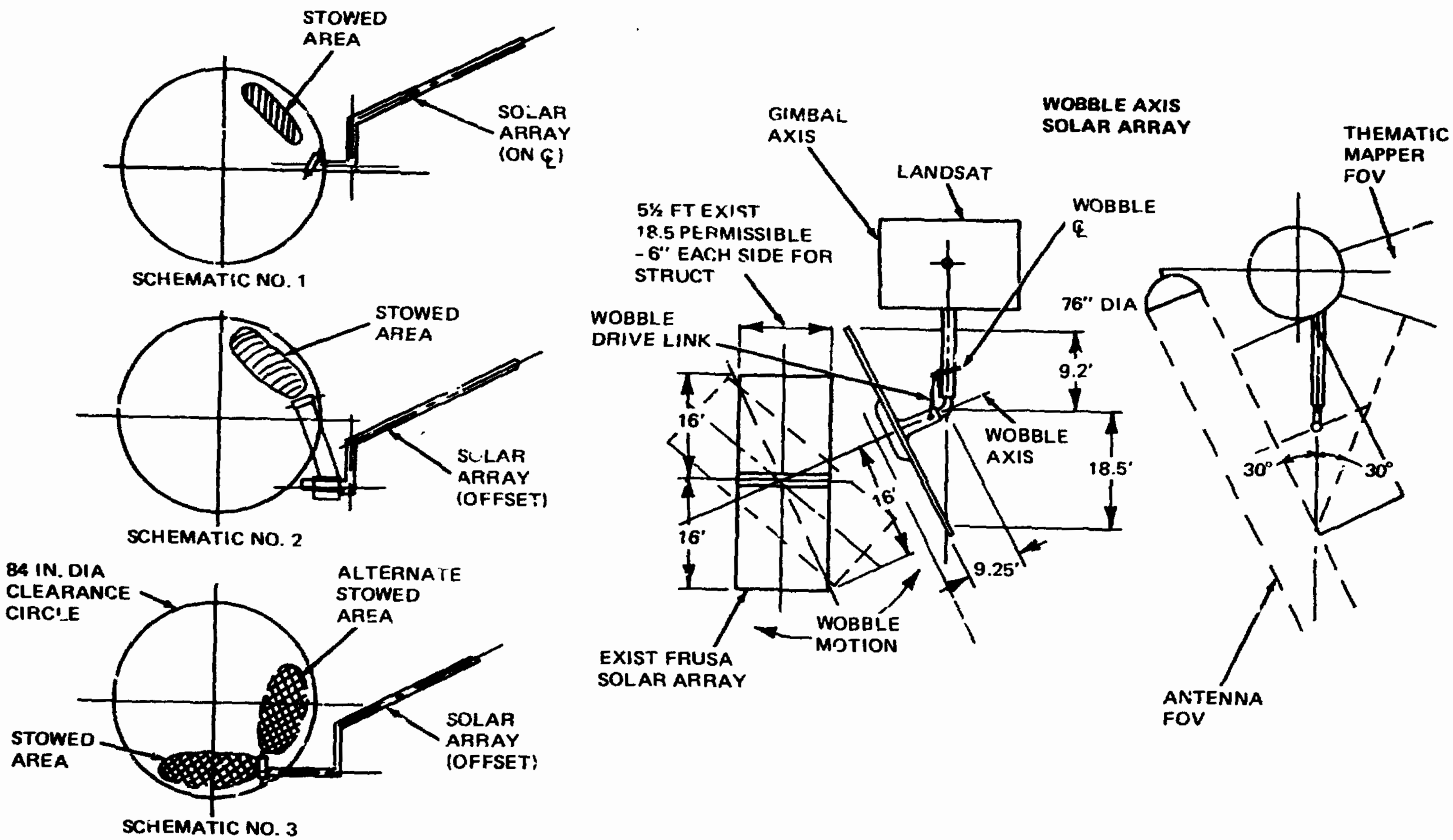
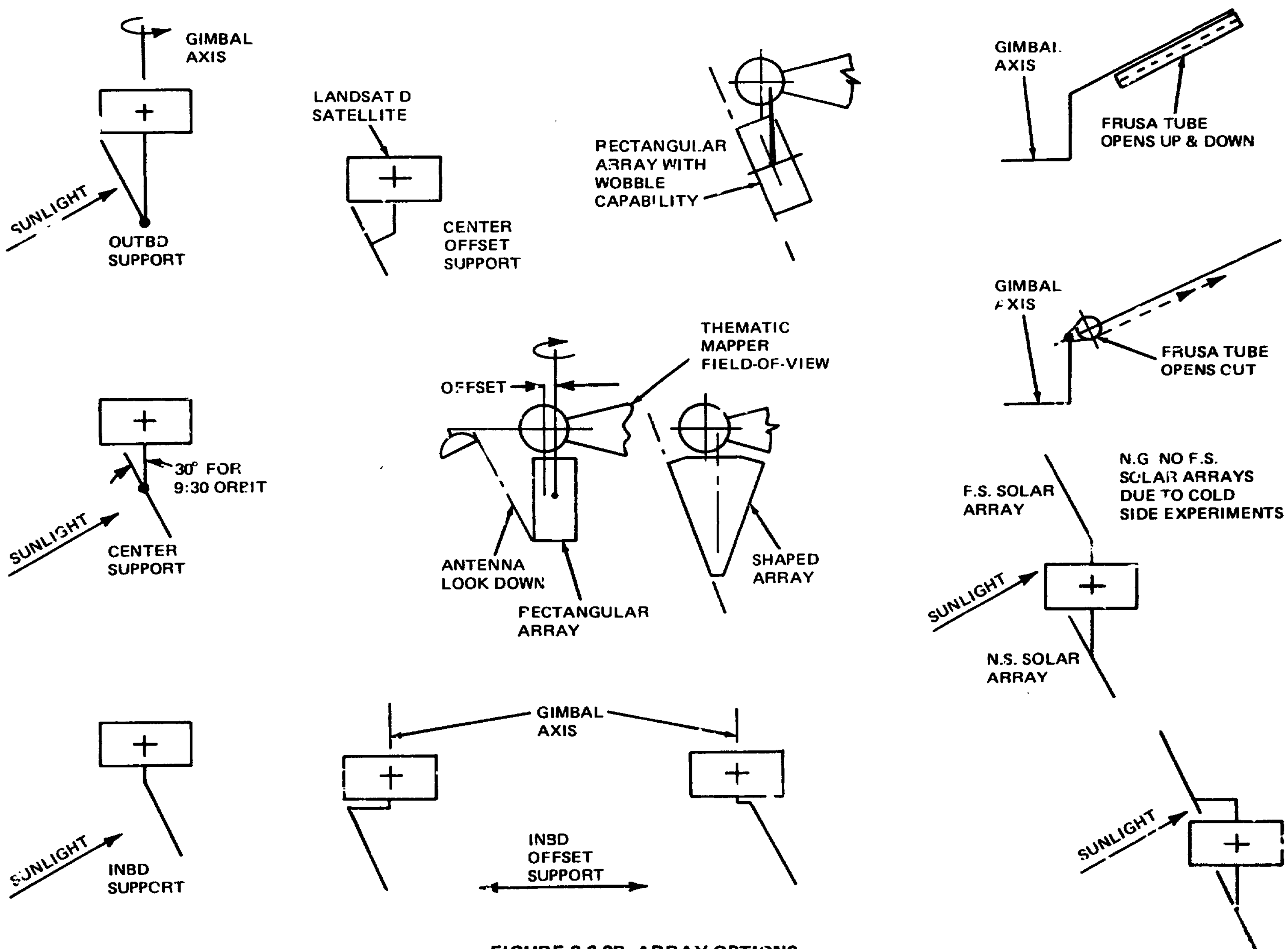
FIGURE 2.6-1G APPENDAGE GEOMETRY



2473-41

FIGURE 2.6-2A ARRAY OPTIONS



2473-41

FIGURE 2.6-23 ARRAY OPTIONS

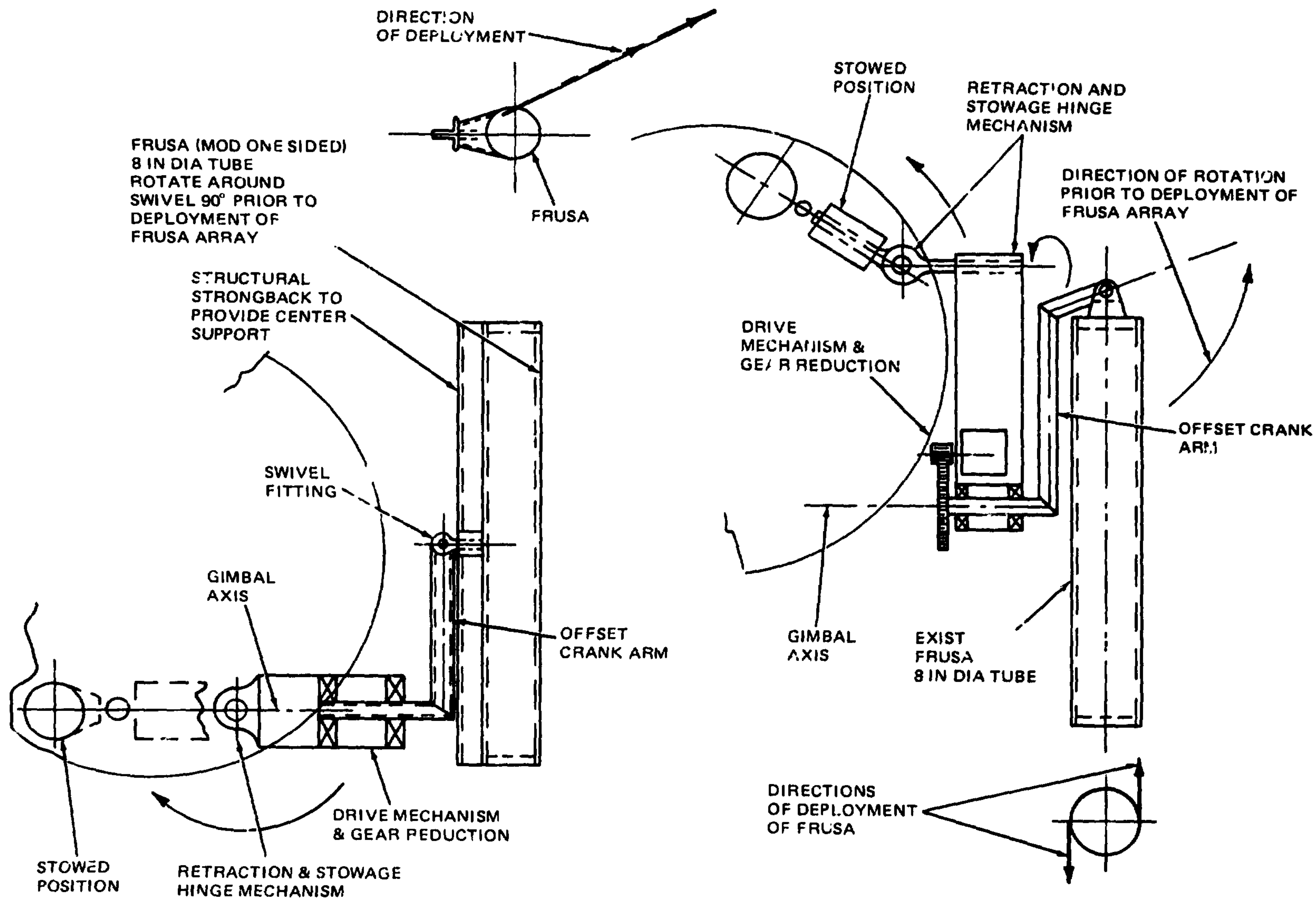
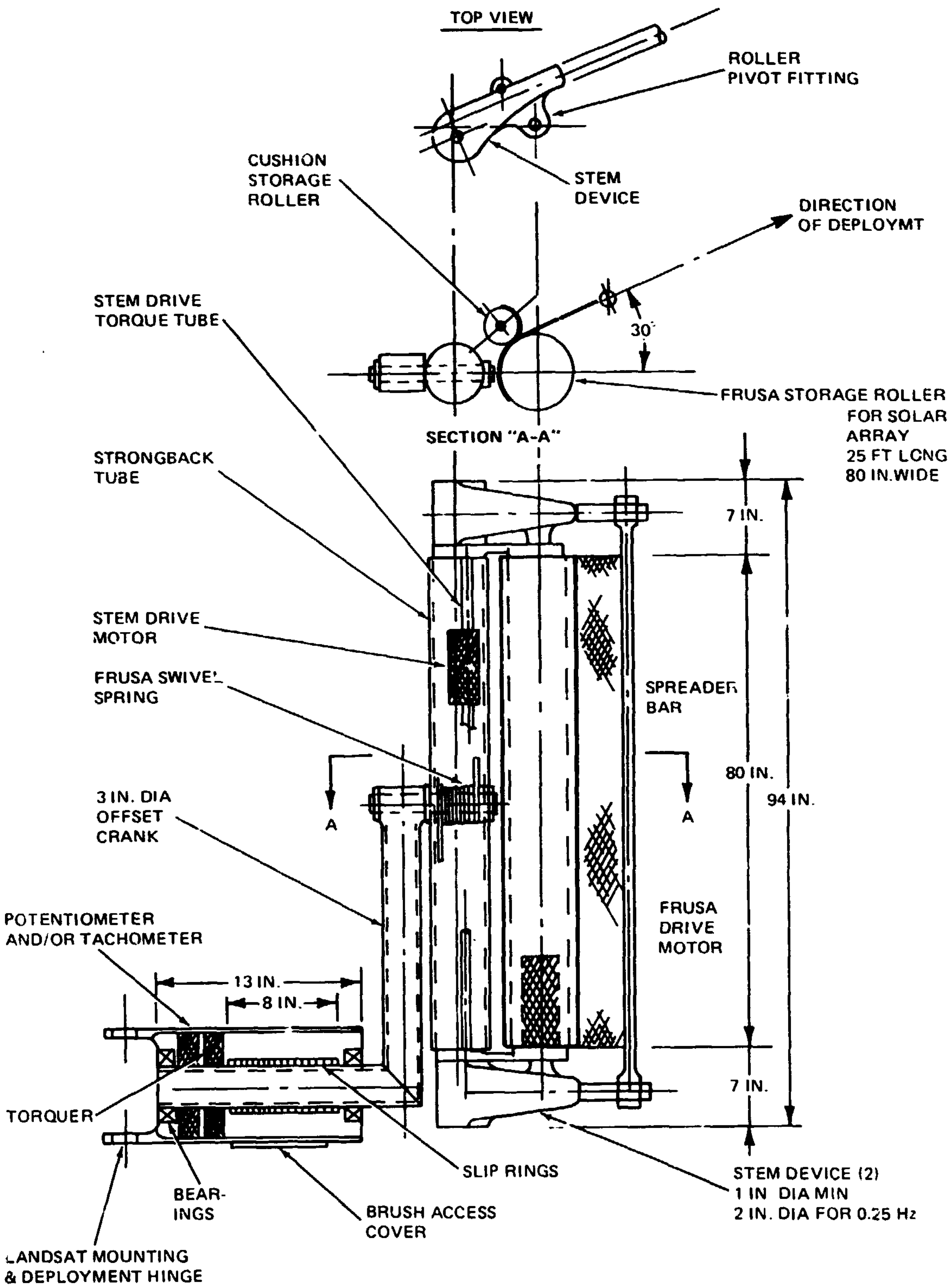


FIGURE 2.6-3A MECHANISM OPTIONS



2473-42

FIGURE 2.6-3B MECHANISM OPTIONS

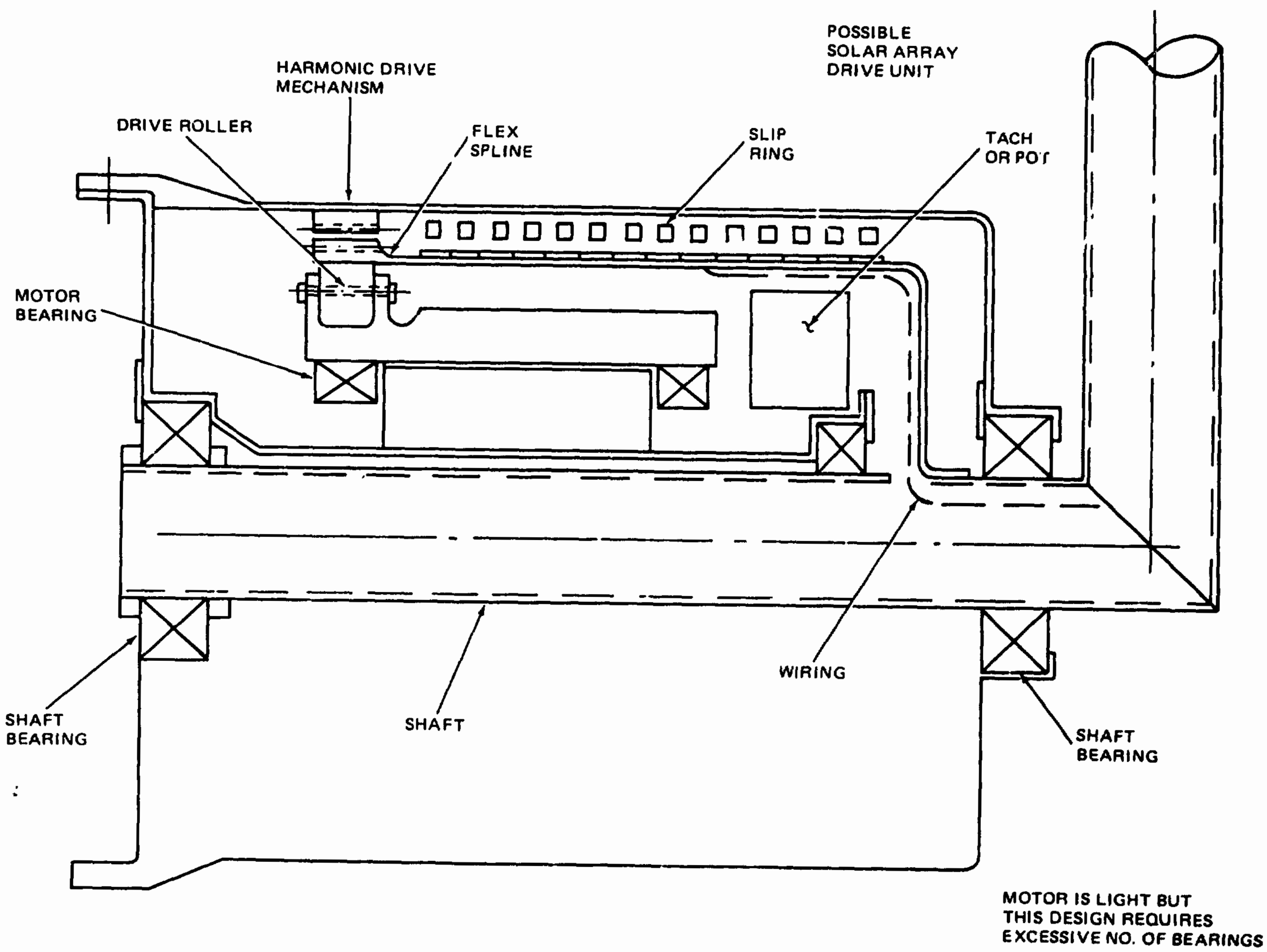
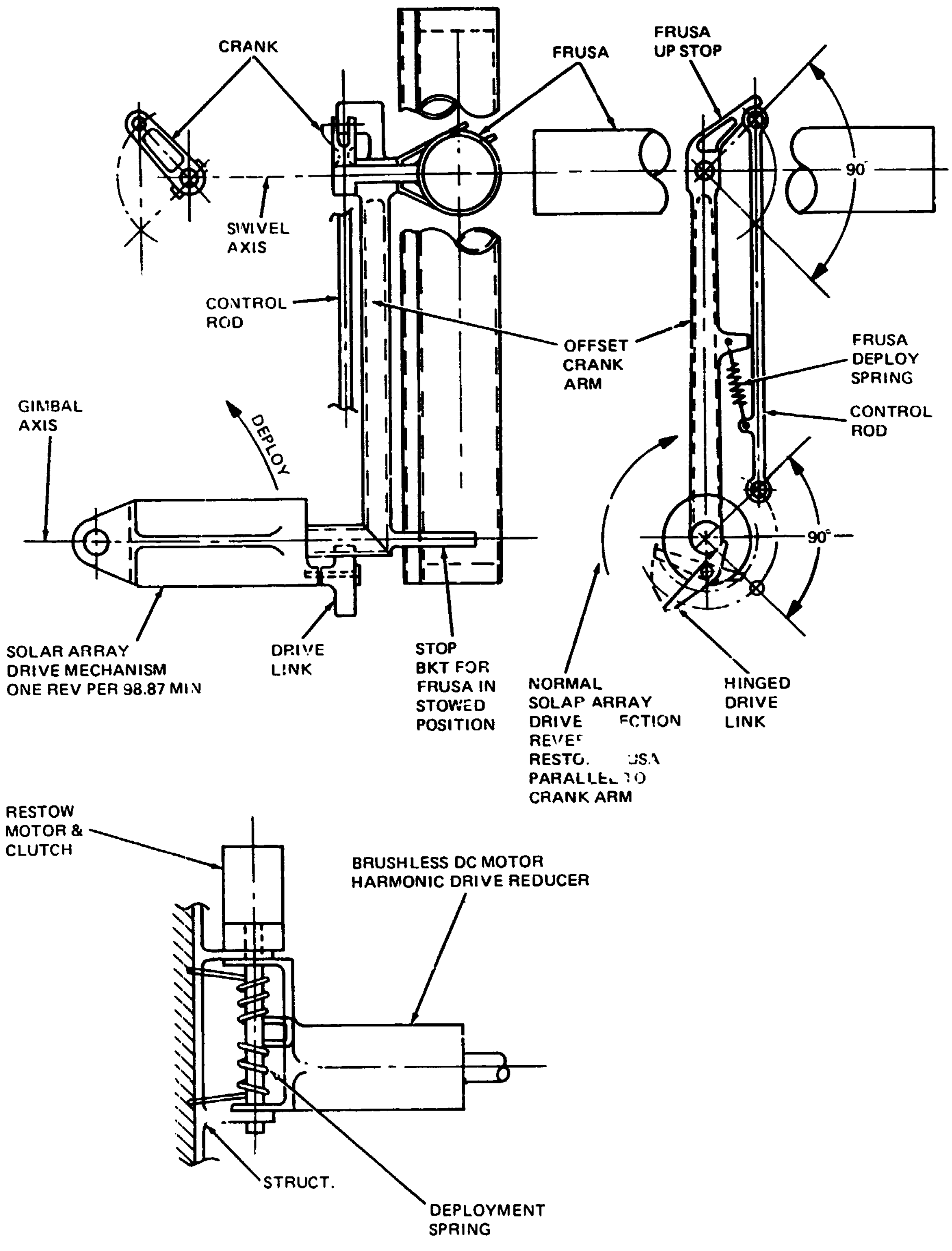
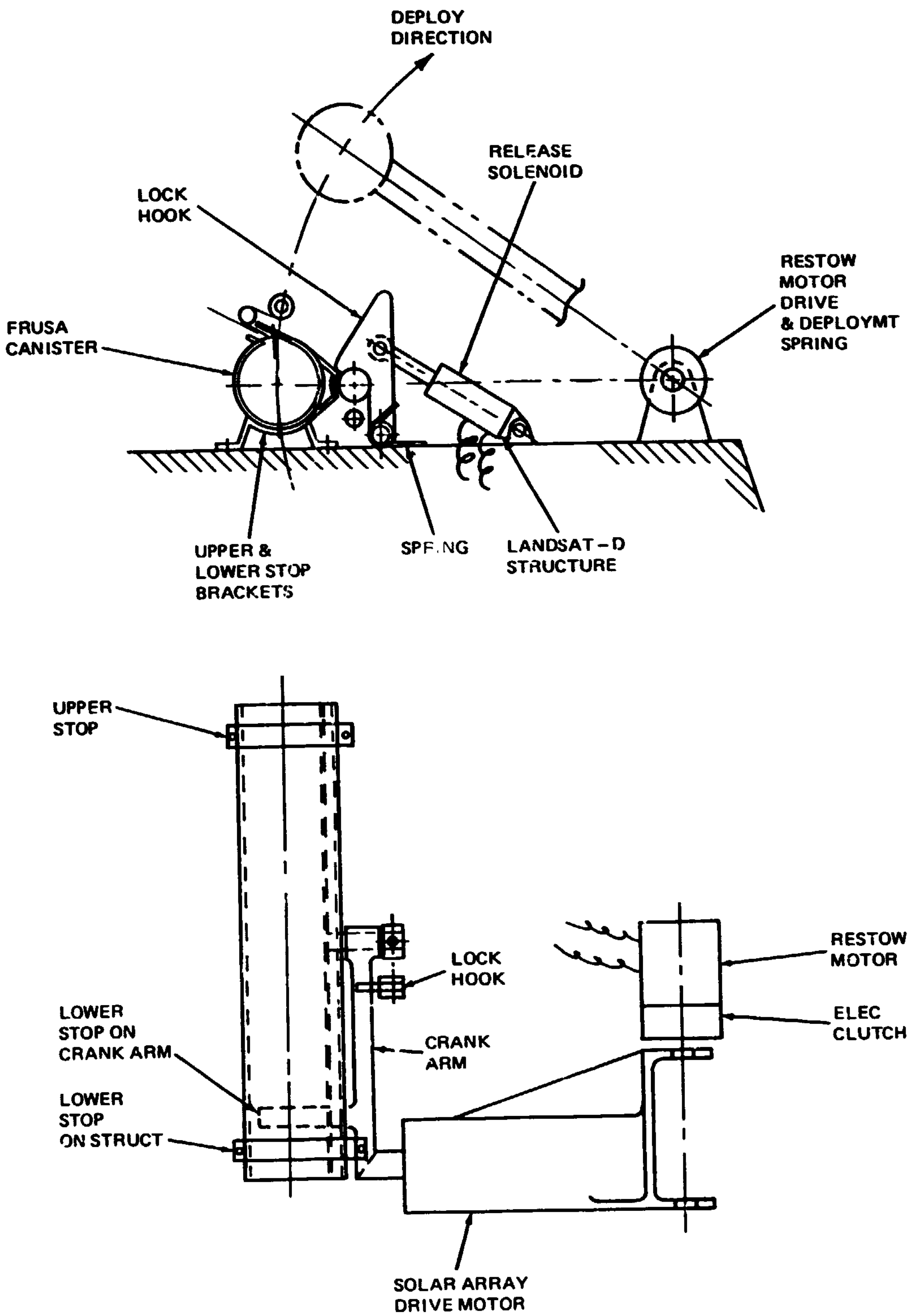


FIGURE 2.6-3C MECHANISM OPTIONS



2473-43

FIGURE 2.6-3D MECHANISM OPTIONS



2473-43

FIGURE 2.6-3E MECHANISM OPTIONS

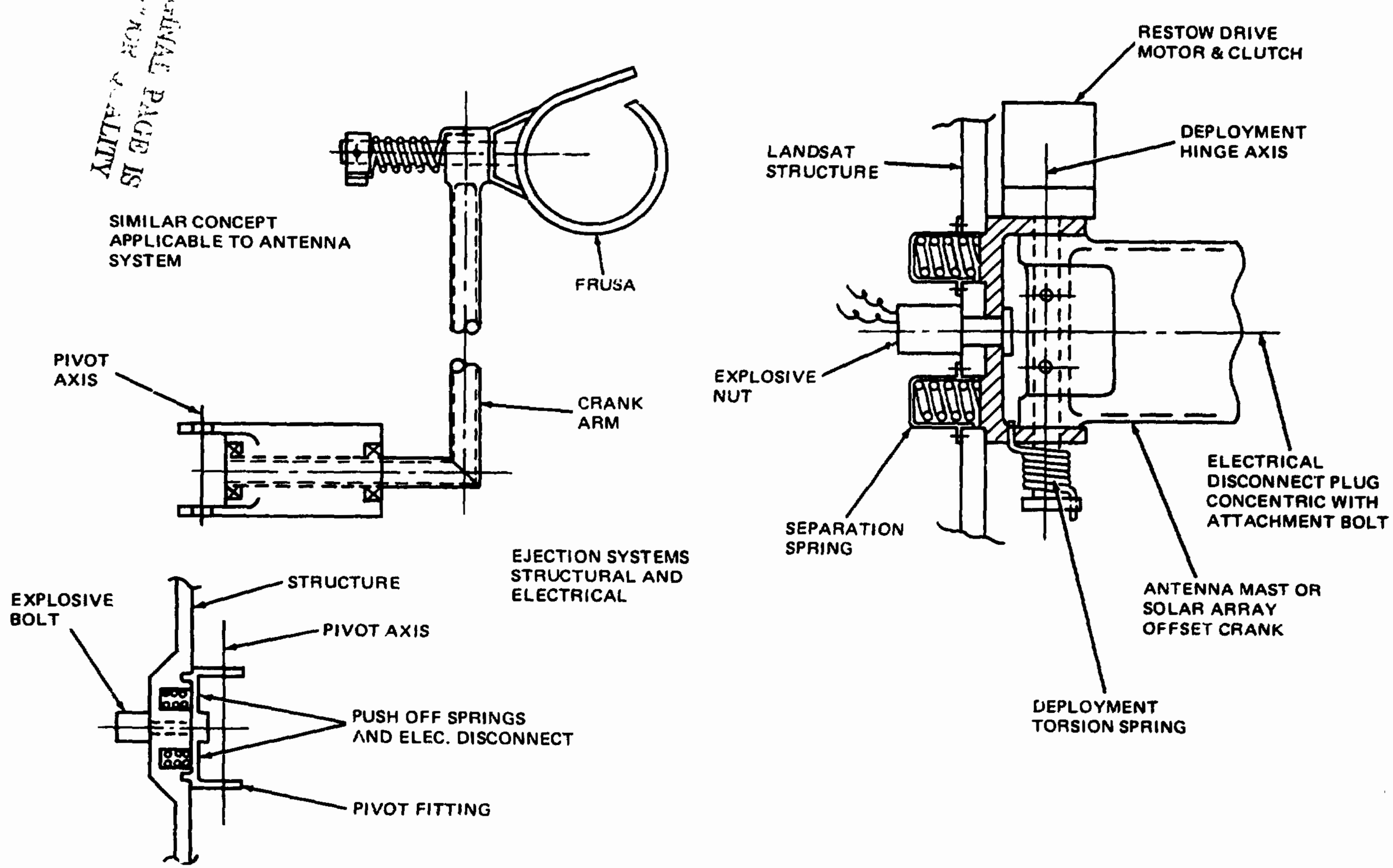


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FIGURE 2.6-3F MECHANISM OPTIONS

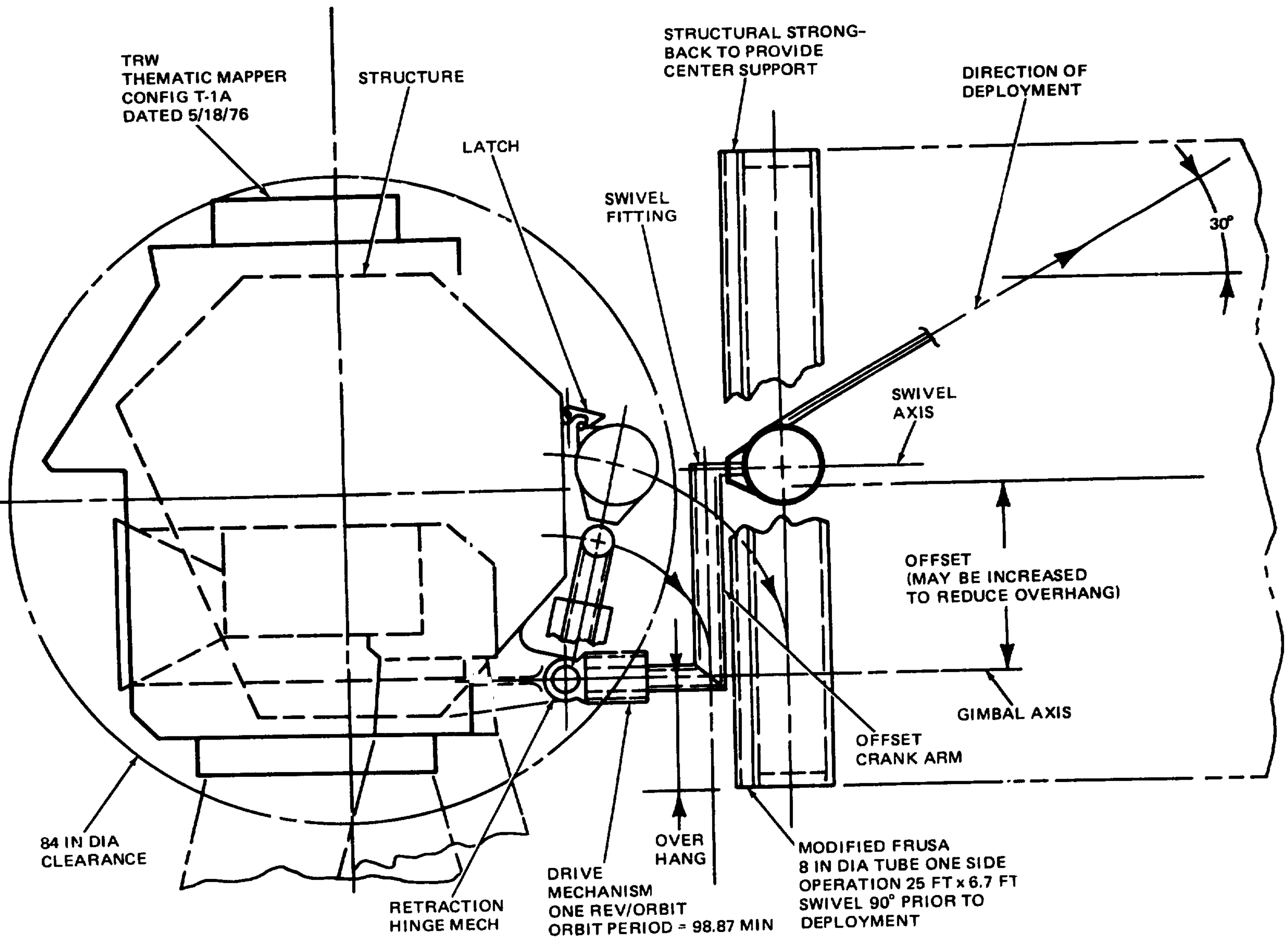

FIGURE 2.6-4 ALTERNATE ARRAY CONFIGURATION I

2473-46

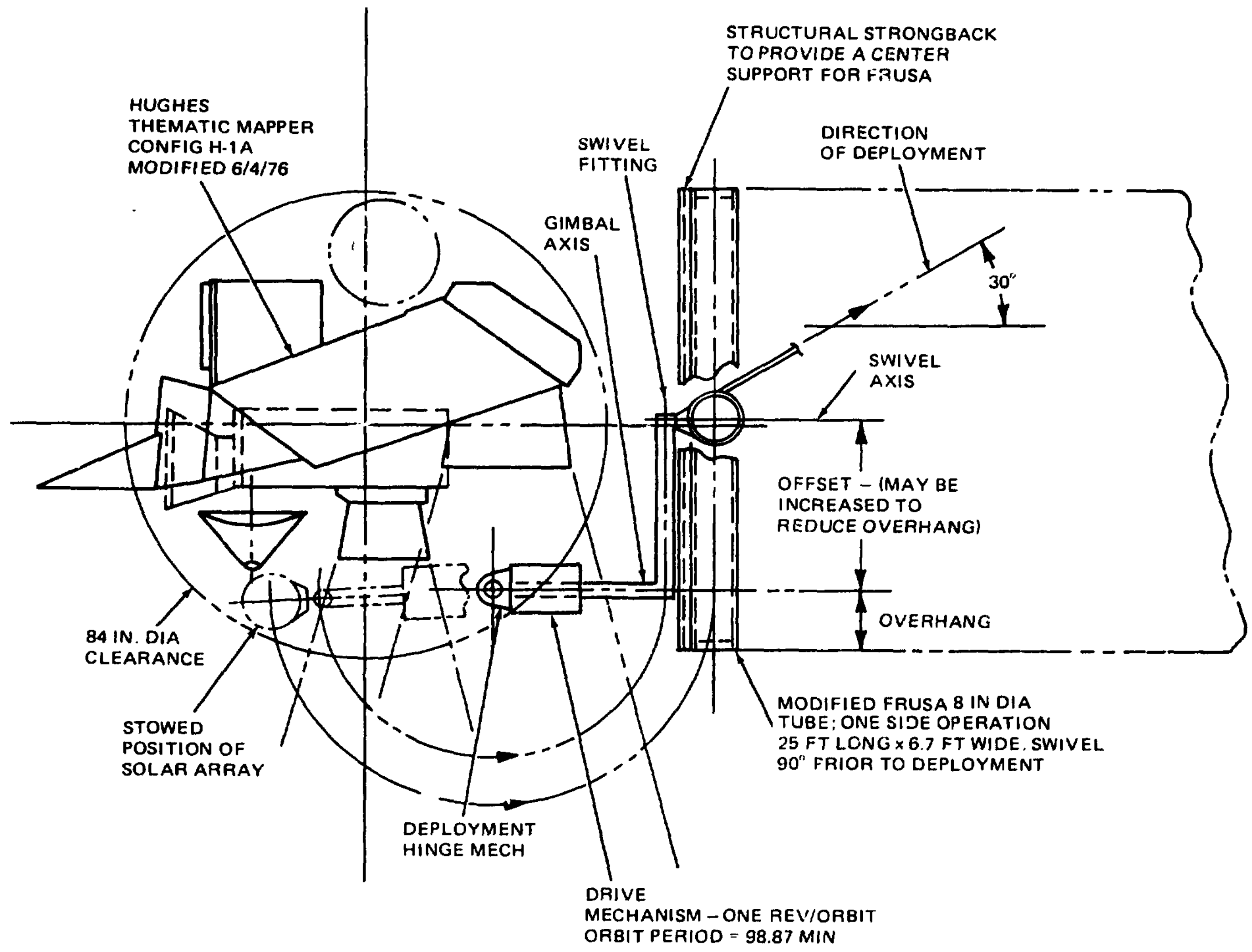

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FIGURE 2.6-5 ALTERNATE ARRAY CONFIGURATION 2

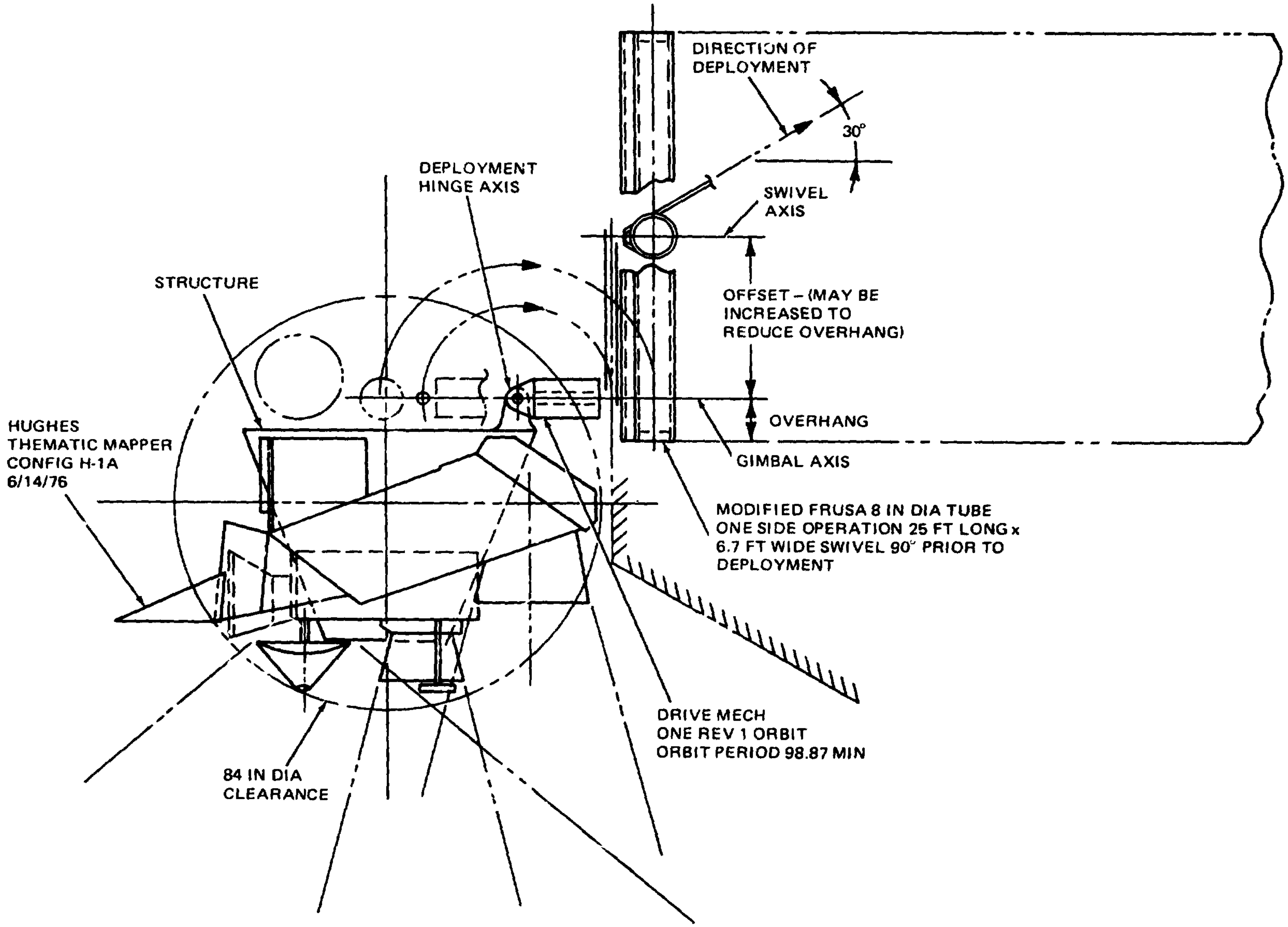

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FIGURE 2.6-6 ALTERNATE ARRAY CONFIGURATION 3

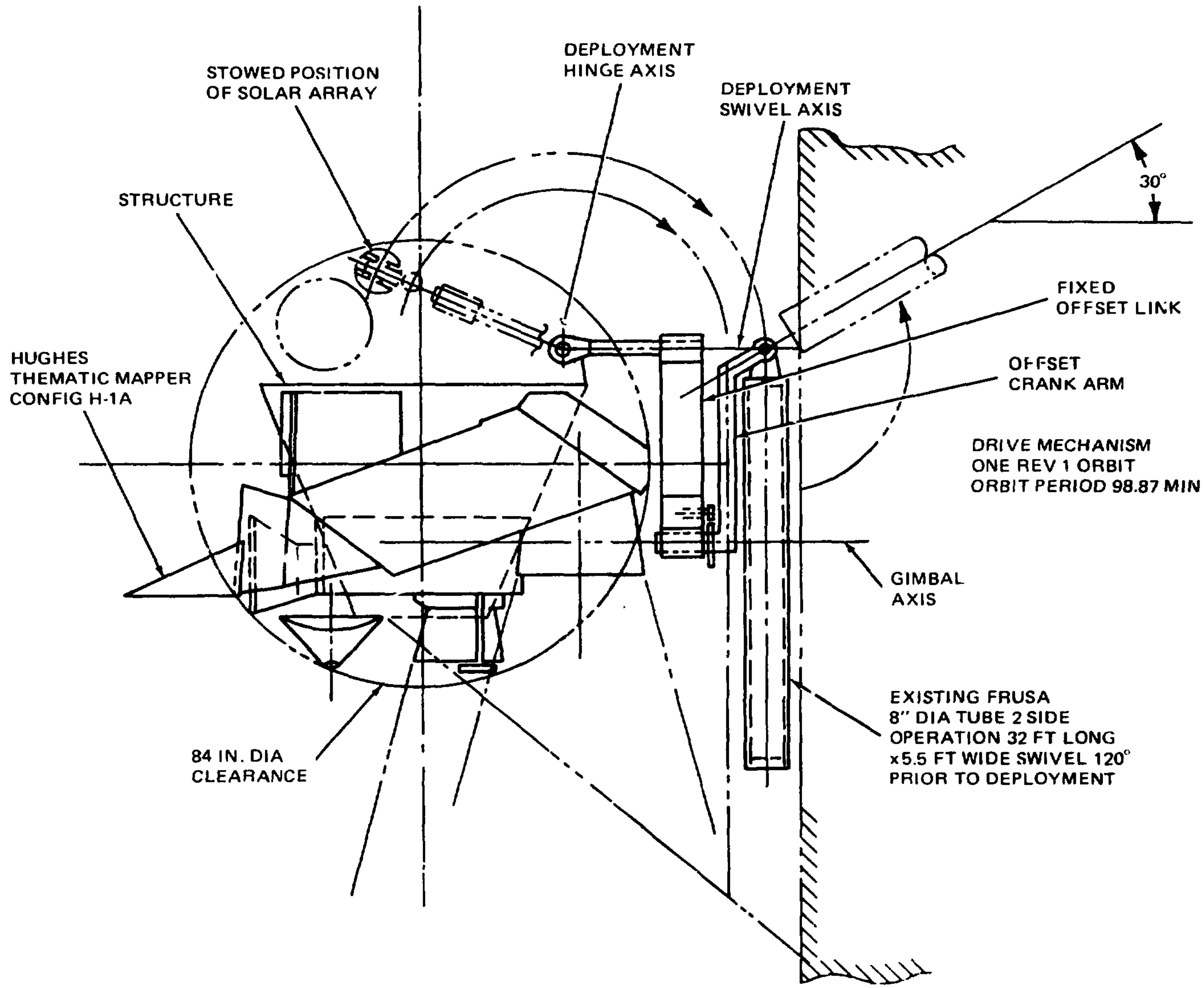

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FIGURE 2.6-7 ALTERNATE ARRAY CONFIGURATION 4

- Depth of base bulkhead - a key stiffness requirement
- Thermal surfaces
- Orientation of the MMS
- Re-orientation in orbit
- Fields-of-View
- Radiator positions
- Packaging efficiency
- Structural flexibility
- Position of stowed appendages
- Complexity of appendage deployments

The evaluation assessment was made fundamentally as a qualitative judgement to the given configuration's potential for further in-depth study. Tables 2.7-1 and 2.7-2 are structural and thermal evaluation tables of the eight broad configuration candidates respectively.

The results of this broad evaluation indicated that the four Class One schemes had the most potential for continued study. Figures 2.7-1 through 2.7-4 show the stowed arrangements for these configurations.

Narrowing down to Class One candidates, a more specific evaluation was made. Tables 2.7-3 and 2.7-4 illustrate the evaluation comments for the structural and thermal disciplines.

As a result of the above described assessments, two candidates, H-1A and T-1A, one for each Thematic Mapper were chosen to be the subject of an in-depth technical verification study - Phase II. Table 2.7-5 is an overall tabulation of the evaluation and ranking of the eight configurations considered. Figures 2.7-5 and 2.7-6 are illustrations of the chosen configurations in their deployed orbital mode.

The selected designs offered the most potential. Some of the more desirable potential features of these configurations (still to be verified in Phase II) are listed below.

- Efficient Structure
- Base Bulkhead offers good continuity to MMS
- Low Weight
- Good access for refurbishment

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TABLE 2.7-1 LANDSAT – INITIAL STRUCTURAL ASSESSMENT

	T-1A	T-1B	T 2	T-3	H-1A	H-1B	H-2	H-3
Length of Structure	43.5 in	66.5 in.	–	99 in	40 in.	33.5 in.	75 in.	39 in.
Load Path to 3 MMS Mounting Pts	Long Cantilever on One Leg	Direct Mounting on Shallow Beam Adapter Works Hard	Long Cantilever on Two Legs	Long Cantilever on Two Legs	Long Cantilever on One Leg	Direct Mounting on Shallow Beam	Short Cantilever on One Leg	Long Cantilever on Two Legs Adapter Works Hard
Relative Stiffness of Config Thrust Axis	Stiff	Stiff	–	Very Flexible	Stiff	Very Flexible	Stiff	Very Flexible
Relative Stiffness of Config-Lateral Axis	Flexible	Flexible	–	Very Flexible	Flexible	Very Flexible	Flexible	Flexible
Structure Mounting of (for Thrust Load): TM (H 250 & T 650 lb)	Direct Mounting on Deep Beam & Col	Direct Mounting on Shallow Beam	–	Cantilever Mounting on Shallow Beam	Direct Mounting on Deep Beam	Direct Mounting on Shallow Beam	Cantilever on Deep Beam	Cantilever on Shallow Beam
Antenna 180 lb	Cantilevered from Base of Antenna	Mounted on Base with Lateral Brace	–	No Room for Antenna	Cantilevered from Base of Antenna	Cantilevered from Base of Antenna	Mounted on Base with Good Support	Cantilevered from Base of Antenna
Solar Array 150 lb	Cantilevered from Deep Beam	Cantilevered from Deep Beam	–	Cantilevered from Deep Beam	Cantilevered from Deep Beam	Cantilevered from Deep Beam	Cantilevered from Deep Beam	Cantilevered from Deep Beam
MSS 148 lb	Cantilevered from Deep Beam – Short Couple	Cantilevered from Torque Box – Short Couple	–	Cantilevered from Deep Beam – Short Couple	Cantilevered from Deep Beam – Short Couple	Direct Mounting on Shallow Beam	Cantilevered from Deep Beam – Long Couple	Direct Mounting on Shallow Beam
WBM 110 lb	Cantilevered from Deep Beam – Short Couple	Cantilevered from Torque Box – Short Couple	–	Cantilevered from Deep Beam – Long Couple	Cantilevered from Deep Beam – Short Couple	Direct Mounting on Shallow Beam	Cantilevered from Deep Beam – Long Couple	Cantilevered from Deep Beam –
Relative Weight	Least Wt	Heavier	Poor	–	Least Wt	Heaviest	Low Wt	Heavier
Structure Preference	1 st	2	–	–	2	4	1 st	3

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- Good equipment mounting
- Small trapped incident radiation
- Clear fields-of-view
- Short TDRSS Antenna requirements
- Conventional aluminum construction box structure with interior space available for secondary equipment
- Good potential for high fundamental structural frequencies
- Potential for a simple - insulation/heater power-thermal system
- Nominal heater power requirements
- A compact stowed arrangement of equipment.

The study having chosen the two most promising Landsat spacecraft configurations, was ready to point toward an in-depth verification of the selected designs.

Section 3, will discuss the results of these investigations.

TABLE 2.7-2 PRELIMINARY THERMAL EVALUATION – LANDSAT CONFIGURATIONS

Configuration/ Criteria	H-1A	T-1A	H-1B	T-1B	H-2	T-2	H-3	T-3
Provision for Radiators	TM & MSS – All Config Satisfactory WBM – Radiator Requirements Not Defined (Area)							
Ease of Thermally Isolating at Struct. Attach	All Config Equal (Based on Present Definition)							
*Area to be Insulated Proportional to Heater Requirements	Moderate	Moderate	High	High	High	High	Moderate	Moderate
*Geometry Probability of Trapping Incident Radiation	Moderate	Moderate	High	High	Moderate	Moderate	Moderate	Moderate
Effects on MMS					Exposes MMS Power Module to Solar Radiation/Albedo			

*Assuming structure includes shear panels –

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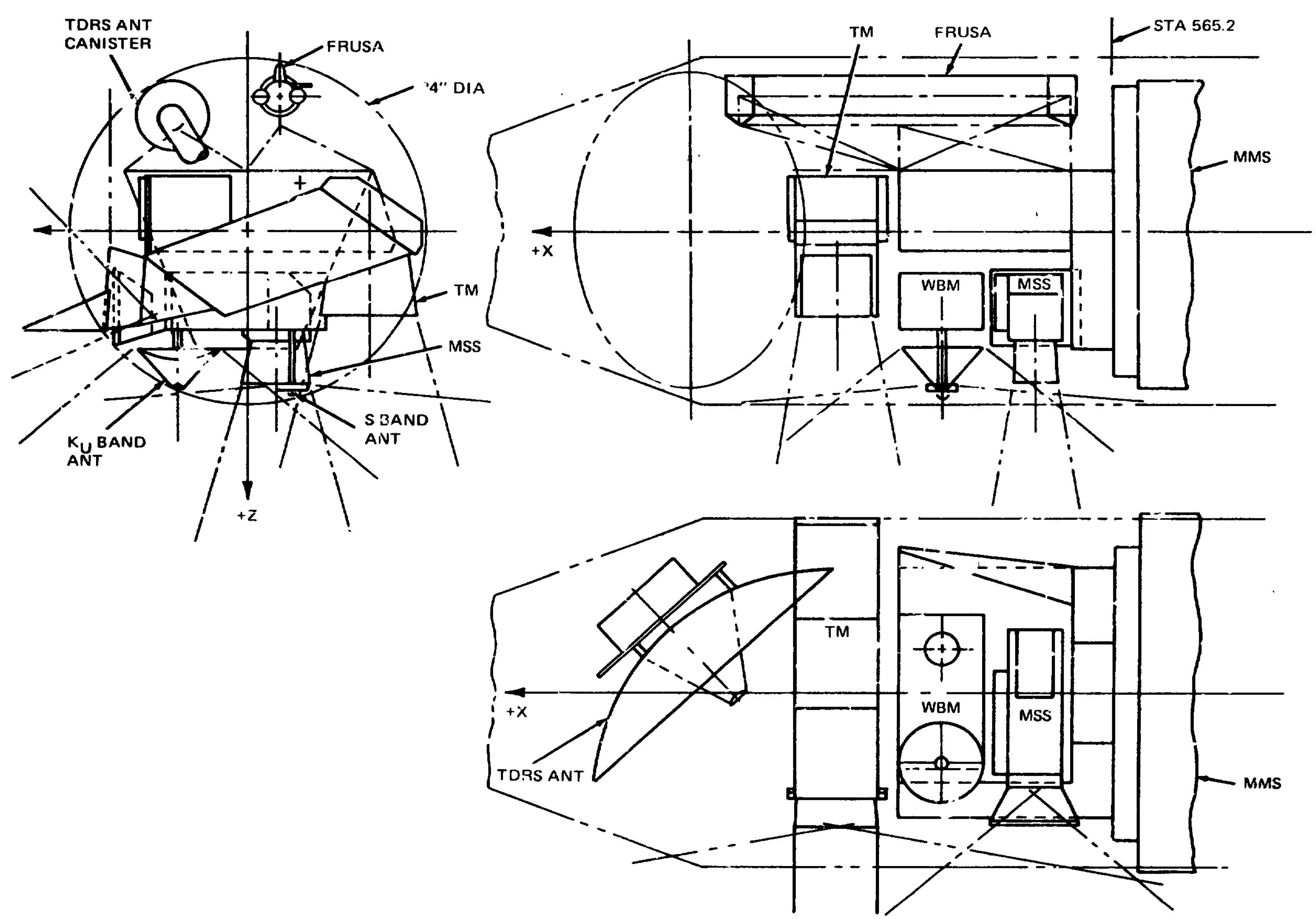

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FIGURE 2.7-1 ARRANGEMENT H-1A

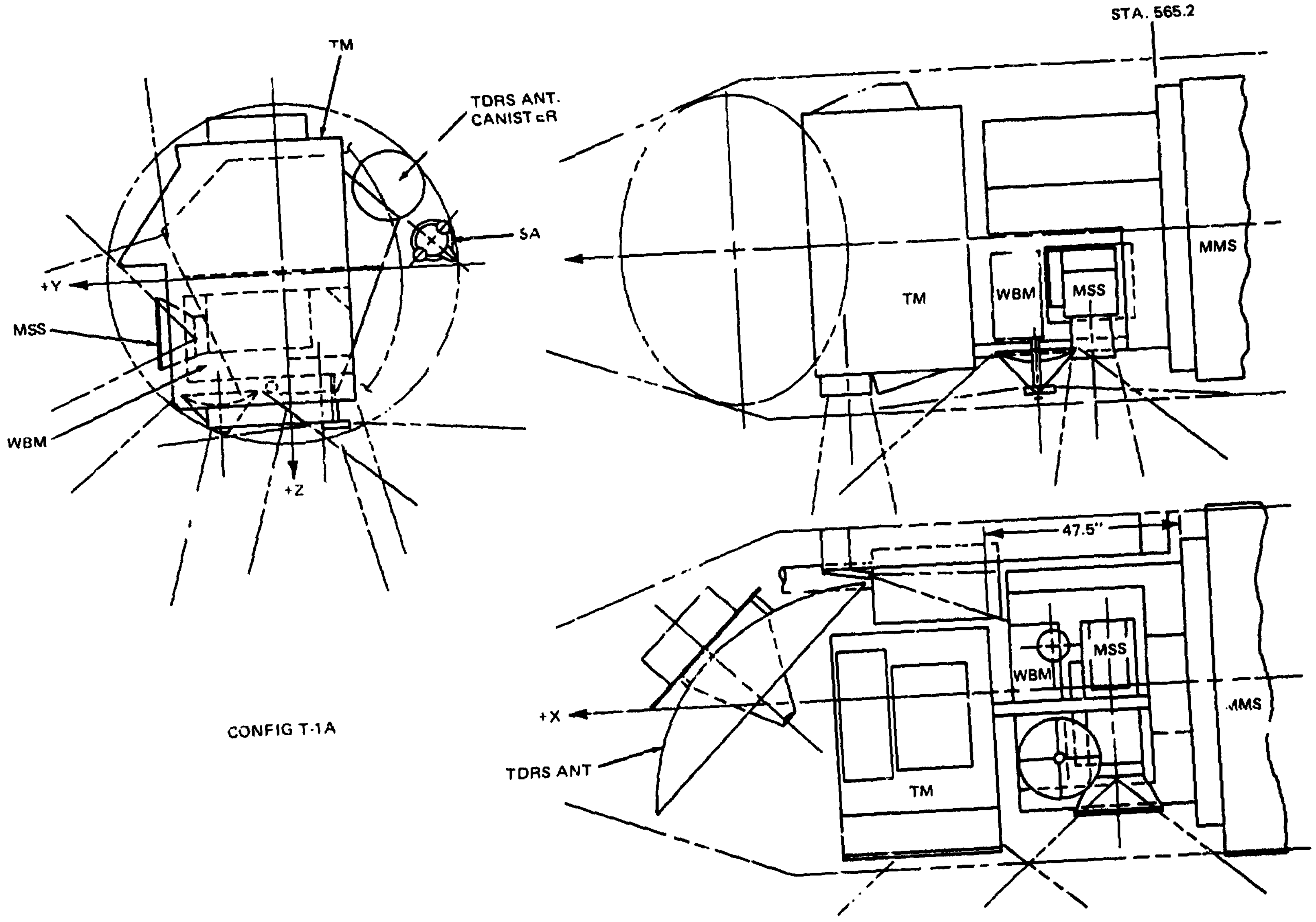

FIGURE 2.7-2 ARRANGEMENT T-1A

2473-55

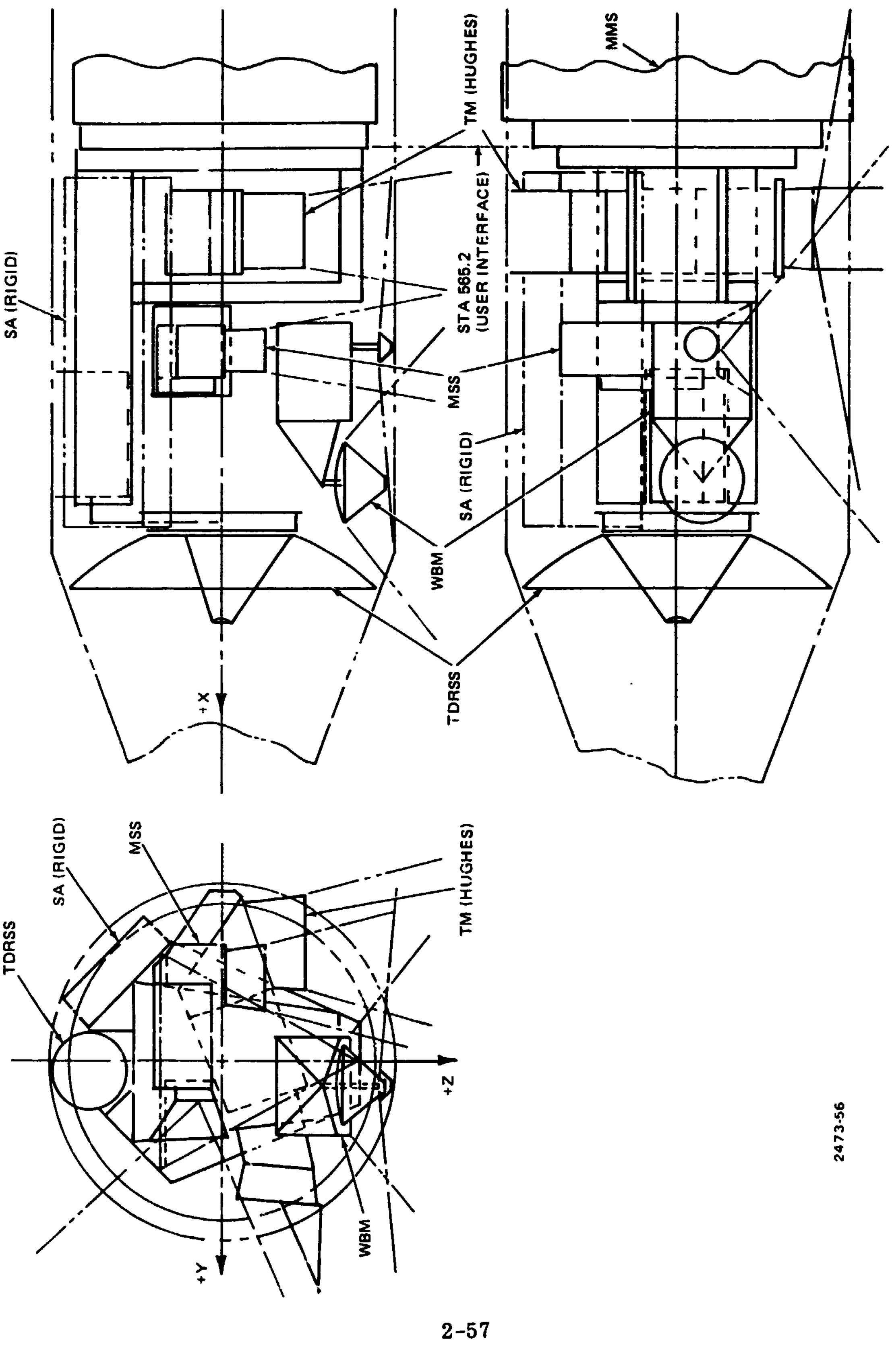

FIGURE 2.7-3 ARRANGEMENT H-1B

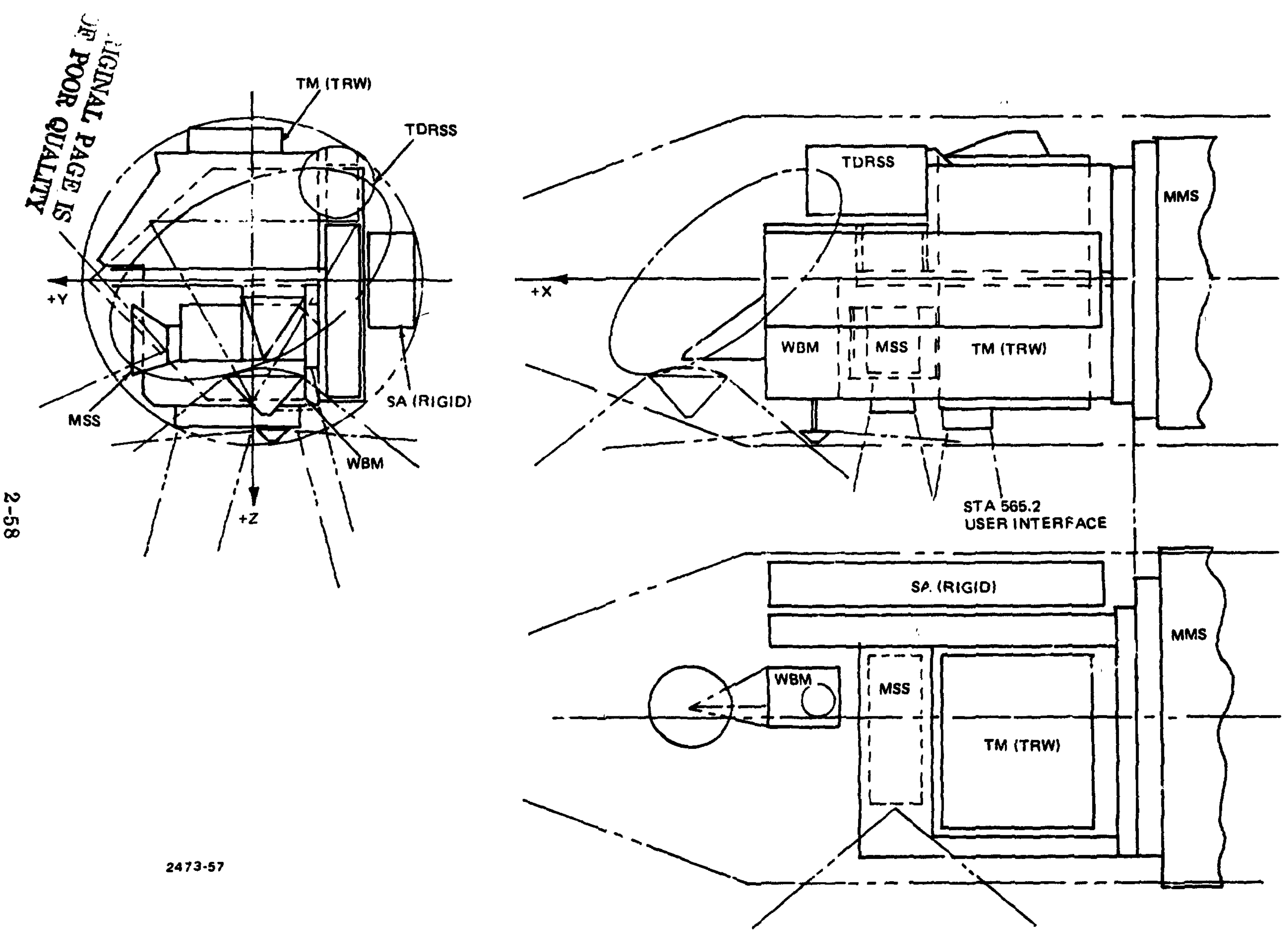

FIGURE 2.7-4 ARRANGEMENT T-1B

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TABLE 2.7-3 CLASS ONE EVALUATION – CONFIGURATION

	H-1A	H-1B	T-1A	T-1B
Primary Structure	++ Rel Stiff Structure ++ Good MMS Cont + Good Commonality + Low Weight	– Flex Arch Struct – Poor – Poor – Heavy	+ Rel Stiff Struct + Good MMS Cont + Good Commonality + Low Weight	– Flex Arch Struct – Poor – Poor – Heavy
Experiments	+ FOV Rqmts Satisfied + Good WBM Rad Growth ++ Good TM Mount + Stiff MSS & WBM Supt	– TM Rad Impinge + Same ++ Same – Flex Supt	+ FOV Rqmts Satisfied – Limited WBM Rad Area + Good TM Mount + Stiff MSS & WBM Supt	+ Same + Good Growth Potent ++ Good TM Mount – Flex Supt
Appendages	++ Good TDRS Stowage + Min TDRS Cantilever ++ Good SA Stowage + Simple Deploy Mech	++ Same ++ Direct CG Mount + Adequate + Same	+ Limited TDRS Stowage + Cantilevered TDRS + Adequate SA Stowage + Simple Deploy Mech	– Marginal + Same + Same + Same
Resupply	++ Good Access to all Experiments	+ Same	+ Hinged S-Band Ant Required	+ Good Access

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TABLE 2.7-4 CLASS ONE EVALUATION – THERMAL

Inertia	H 1A	T-1A	H-1B	T-1B	Preferred
Location of Coolers for Near View to Y Directions	Good	Good	Good	Good	Equal
Suitability of Design for Minimizing Refl/Emitted Energy into Coolers	Poor	Fair	Poor	Fair	T-1A
Provisions for WBM Radiator	Good	Fair	Good	Good	H-1A/T-1B
Ease of Thermally Isolating from Structure	Good	Good	Good	Good	Equal
Probability of Trapping Incident Radiation	Moderate	Moderate	High	High	H-1A
Structure Area to be Insulated	Least	Moderate	Large	Largest	H-1A
Adaptation to Truss Type Structure	Fair	Fair	Poor	Poor	H-1A
Ease of Thermally Isolating from MMS	Good 4 Pts	Poor 7 Pts	Fair 6 Pts	Poor 8 Pts	H-1A
Compatibility with MMS Requirements	Good	Good	Good	Good	Equal

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TABLE 2.7-5 KEY COMPARATIVE COMMENTS/EVALUATION

	H-1A	I-1A	H-1B	T-1B	H-2	T-2	H-3	T-3
	+ Good Struct + Low Weight + Good Portability Good Common + Good MMS Cont + Good TM Mount – Lim Depth Base Bhd + Clean Therm Surf – Higher CG	+ Good Struct + Low Weight + Good Common + Good MMS Cont – TM Overhung – Lim Depth Base Bhd + Clean Therm Surf – Higher CG – Stowed App Lim	– Flex Arch Struct – Heavy – Poor MMS Cont + Good App Stow – High Inc Rad – Limit Port – Low Common + Deep Base Bhd + Good TM MT	– Flex Arch Struct – Heavy – Poor MMS Cont + Good App Stow + Deep Base Bhd – Low Common – Low Port + Good TM Mt – Poor WBM Ant	– Complex Array D – Poor MMS Orient + Good Portability + Good Common + Good MMS Cont + Good Struct – Poor TM Mt + Good App Stow – Reorient Reqd – Array-Thr Int – Higher CG	– Complex Array D – Poor MMS or Cnt – Poor TM Mt – Poor Struct – Poor App Stow – Ineff Pkg Vol – Flex Struct – Impractical – Poor WBM Ant – Reorient Reqd – Array-Thr Int – Low CG	– Poor Cont to MMS – Complex Ant D + Compact Struct – Poor TM Mt + Large Ant Stow – Reorient-Thr + Deep Base Bhd + Low CG – Poor WBM Ant	– Long Struct – Heavy – Flexible – Poor MMS Cont – Poor App Stow – Complex Ant D – Poor TM FOV + Good Port + Good Common – Reorient Thrust – Poor TM Mt
MERIT VALUE	+7 –2	+5 –4	+3 –6	+3 –6	+5 –6	+1 –11	+4 –5	+2 –9
RANK	1	2	5/6	5/6	3/4	8	3/4	7

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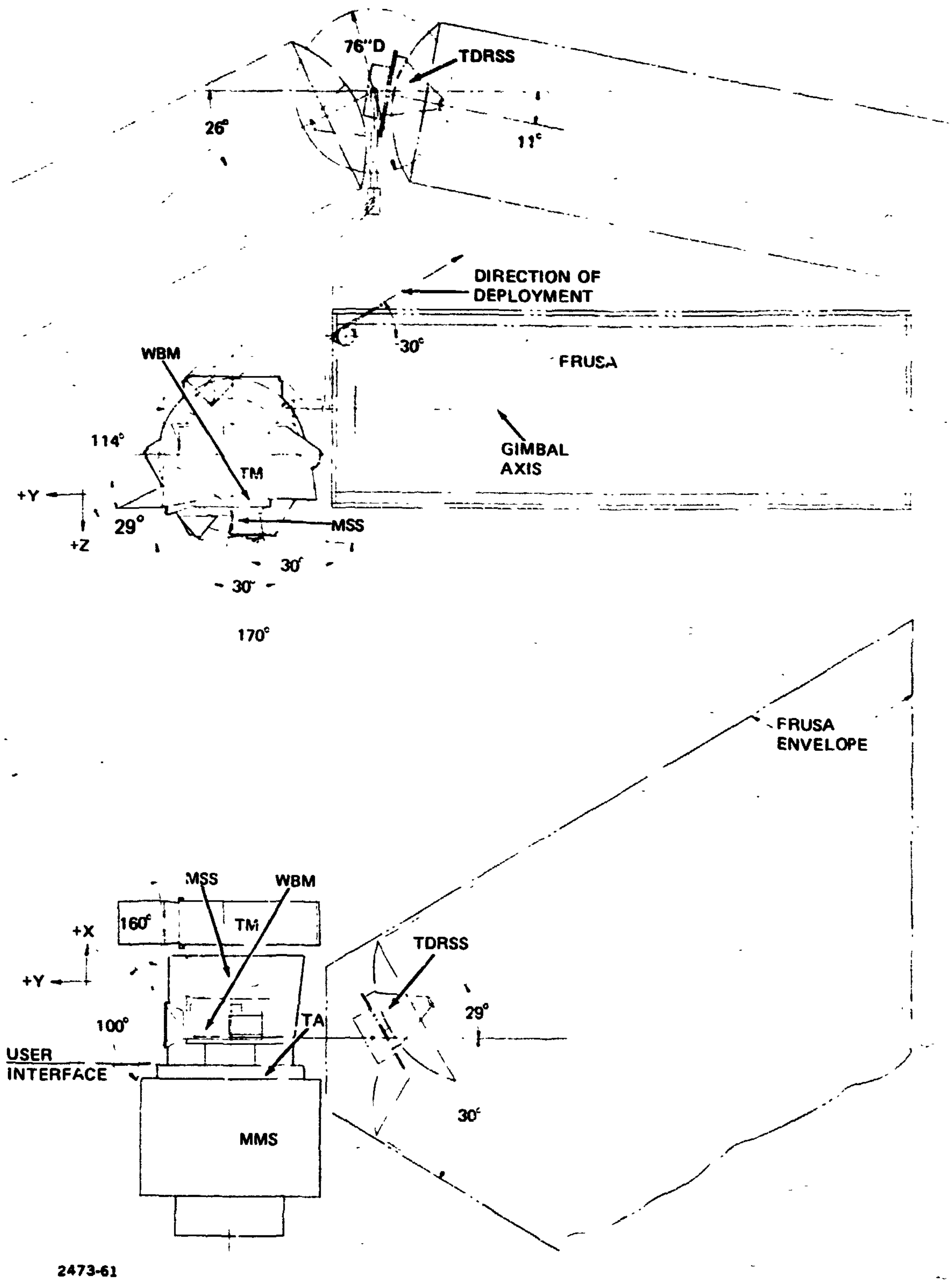


FIGURE 2.7-5 H-1A ORBITAL ARRANGEMENT

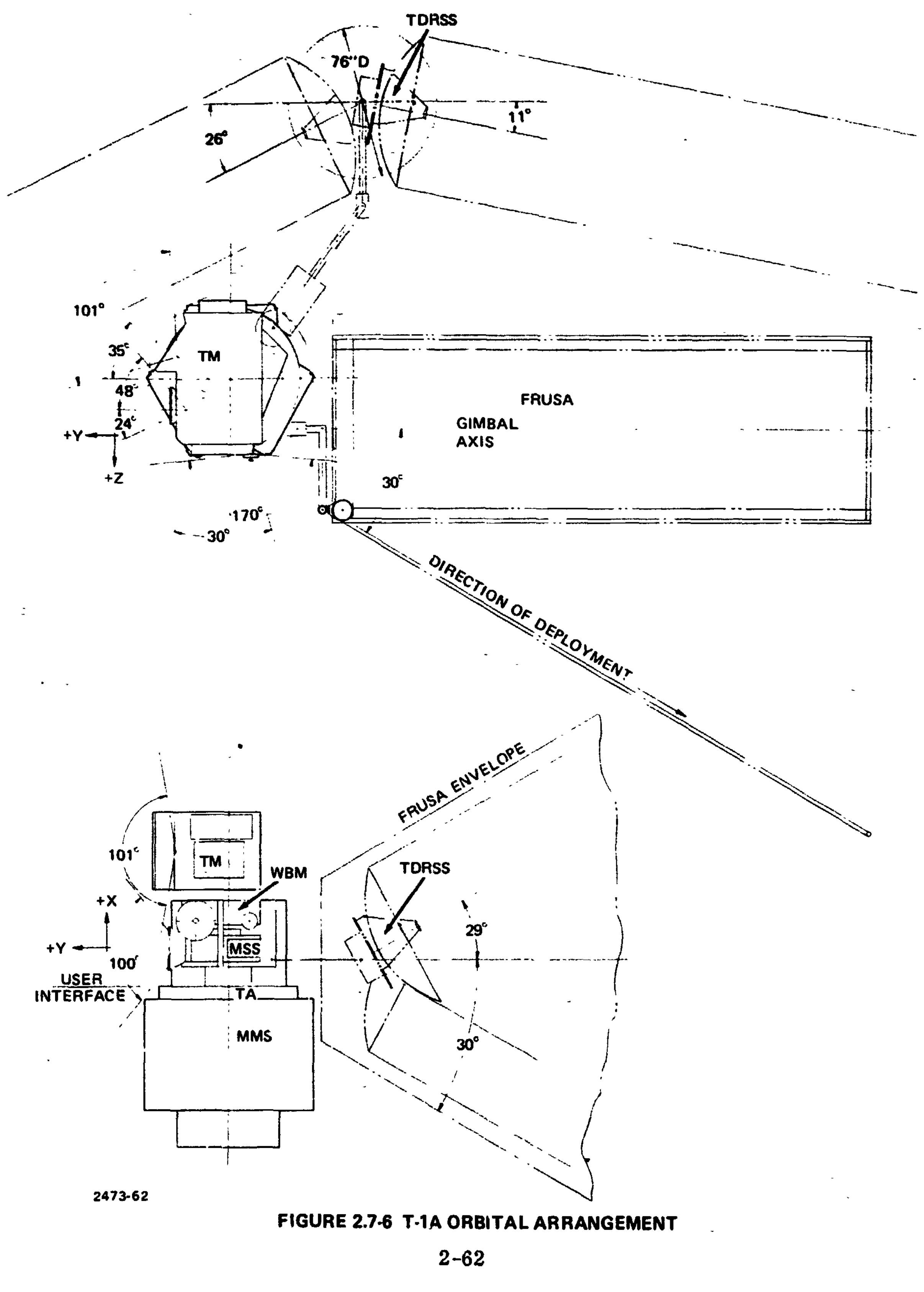


FIGURE 2.7-6 T-1A ORBITAL ARRANGEMENT

SECTION 3

CONCEPT VALIDATION

The Concept Validation portion of the Phase II study is primarily devoted to design investigations of a more detailed nature than those undertaken in Phase I. The purpose of this phase of the study is to develop the design of the selected T-1A and the H-1A configuration to a level sufficient to establish concept feasibility. The results of this investigation indicate conclusively that these two configurations are feasible for the Landsat follow-on spacecraft.

Efforts in mechanical design were mainly directed to the areas of structures, supports, equipment arrangements, mounting, orientation, and deployment mechanisms for the Solar Array and the TDRS antenna. Other design efforts, such as thermal design, involved concept definition, materials selection, and estimates for heater power requirements.

The analytics of structural design, environmental dynamics, and thermal design were all aided by the use of math models. The structures mathematical model based on a finite element approach was developed to determine load distributions, stresses, and flexibility constants. This structures model also provided the influence coefficients used as input to a subsequent dynamic math model to assess the environmental effects of vehicle launch and orbital operation. Flexible modes and fundamental frequencies for the T-1A stowed and deployed configuration were determined.

In the thermal analyses, orbital heat fluxes were determined by the use of a generalized model consisting of thirty surfaces to simulate the Instrument Module configuration. Based upon these results and simplified thermal analyses, a preliminary thermal design consisting of insulation blankets and electric heaters were evaluated.

Mass properties of the integrated Instrument Module were determined by combining NASA/GSFC furnished data for components and data derived from the Grumman evolved designs.

For ease of reference, a summary of interface items was prepared as a separate subsection. Drawings, applicable specifications, and brief discussions of the more significant interface items are included.

Phase II Concept Validation details are presented in the following subsections.

3.1 STRUCTURAL CONFIGURATION

3.2 MECHANICAL SYSTEMS

3.3 STRUCTURAL ANALYSIS

3.4 DYNAMIC ANALYSIS

3.5 THERMAL ANALYSIS

3.6 MASS PROPERTIES

3.7 INTERFACE ITEMS

3.1 STRUCTURAL CONFIGURATION

The subsystem design tradeoffs presented in Phase I have been used to establish the Instrument Module baseline configurations incorporating the Hughes Thematic Mapper and the TRW Thematic Mapper. These configurations satisfy all of the Landsat follow-on spacecraft requirements as presented in the Statement of Work and subsequent discussions with NASA/GSFC. These are not intended to be final configurations, but the two selected for further investigation represent realistic designs upon which concept feasibility can be evaluated.

3.1.1 Selected Configurations

The following objectives summarize the more critical parameters that were used in the development of the selected Instrument Module configurations:

- Satisfy FOV and orbital orientation requirements of the major components comprising the Landsat follow-on spacecraft as described in Subsection 1.2.1 and Subsection 2.7
- Provide adequate clearance with respect to the vehicle fairing dynamic envelope.
- Satisfy access requirements for on-orbit space vehicle resupply
- Design the structure to have a natural frequency of 10 to 15 Hz laterally and 35 Hz axially.

The spacecrafts were configured as compact as practicable, fully utilizing the available payload envelope. More importantly, the selected design produced a very efficient structure because the sizes of the members required for strength closely approached the sizes required for stiffness.

The H-1A and T-1A launch configurations (Refer to Figures 2.7-1 and 2.7-2) show the arrangement of the major components within the fairing dynamic envelope. The TM, MSS and WBM are stacked along the X-axis to satisfy the +Z and +Y view to space requirements. The T[illegible]I is located at the forward end of the arrangement to allow for a short compact structure. The TDRS antenna and Flexible Roll-Up Solar Array (FRUSA) are located in the remaining envelope, near their deployed positions, and clear of the MEMS clearance envelope requirements. Refer to Figure 1.2-7 for an isometric illustration of the H-1A and T-1A stowed configurations.

The H-1A and T-1A on-orbit configurations are shown in Figures 3.1-1 and 3.1-2 respectively. In these configurations, the TDRS antenna and the FRUSA are shown in the deployed position with the limits of operational excursion defined. The Thematic Mapper and other equipment orientations and fields-of-view are also defined in these two figures.

Refer to Figures 1.2-5 and 1.2-6 for equipment installation and removal for the H-1A and T-1A configurations respectively. In the H-1A configuration, the removal direction for MSS and WBM resupply is parallel to the +Y axis, and for the TM, it is parallel to the -Y axis. In the T-1A configuration, the removal direction for the TM and WBM is parallel to the +Y axis, while the removal direction for the MSS is parallel to the -Y axis. The WBM requires an extendable or hinged S-Band antenna mast to clear the vertical TM support post during removal of the TM.

3.1.2 Structures Subsystem

The structural arrangement and details for the H-1A configuration are shown in Figures 3.1-3A and -3B, while Figures 3.1-4A and -4B show the arrangement and details for the T-1A configuration. These primary structures have been analyzed and sized to satisfy the preliminary load factors stated in the supporting analysis of Subsection 3.3. In general, the structural design used large factors of safety (>1.5) and carries uniform structural members extending away from critical regions. Although this approach produces some weight penalty, it offers offsetting advantages in reducing fabrication costs and analysis time. The preliminary structural analysis to-date has resulted in the member sized indicated in Figure 3.1-4. Local load conditions and practical stiffener spacing may result in member sizes different than those generated from the idealized math model.

The T-1A structural arrangement was used to generate the math model geometry for the support analyses discussed in Subsection 3.3. The T-1A configuration was chosen for modeling because of the heavier TM and the narrower vertical structure supporting the other components of the Instrument Module.

The primary structure is basically an L-shaped box structure comprised of a vertical closed torque box of sheet and stringer design set within a triangular shaped 10-in. deep bulkhead. The corners of the bulkhead are mounted to those bolts on the transition ring adapter that straddle the three main longerons of the MSS. The triangular bulkhead geometry is the same for both the T-1A and H-1A, whereas the vertical torque box geometry is dependent on the location of the interface fittings for the two Thematic Mappers.

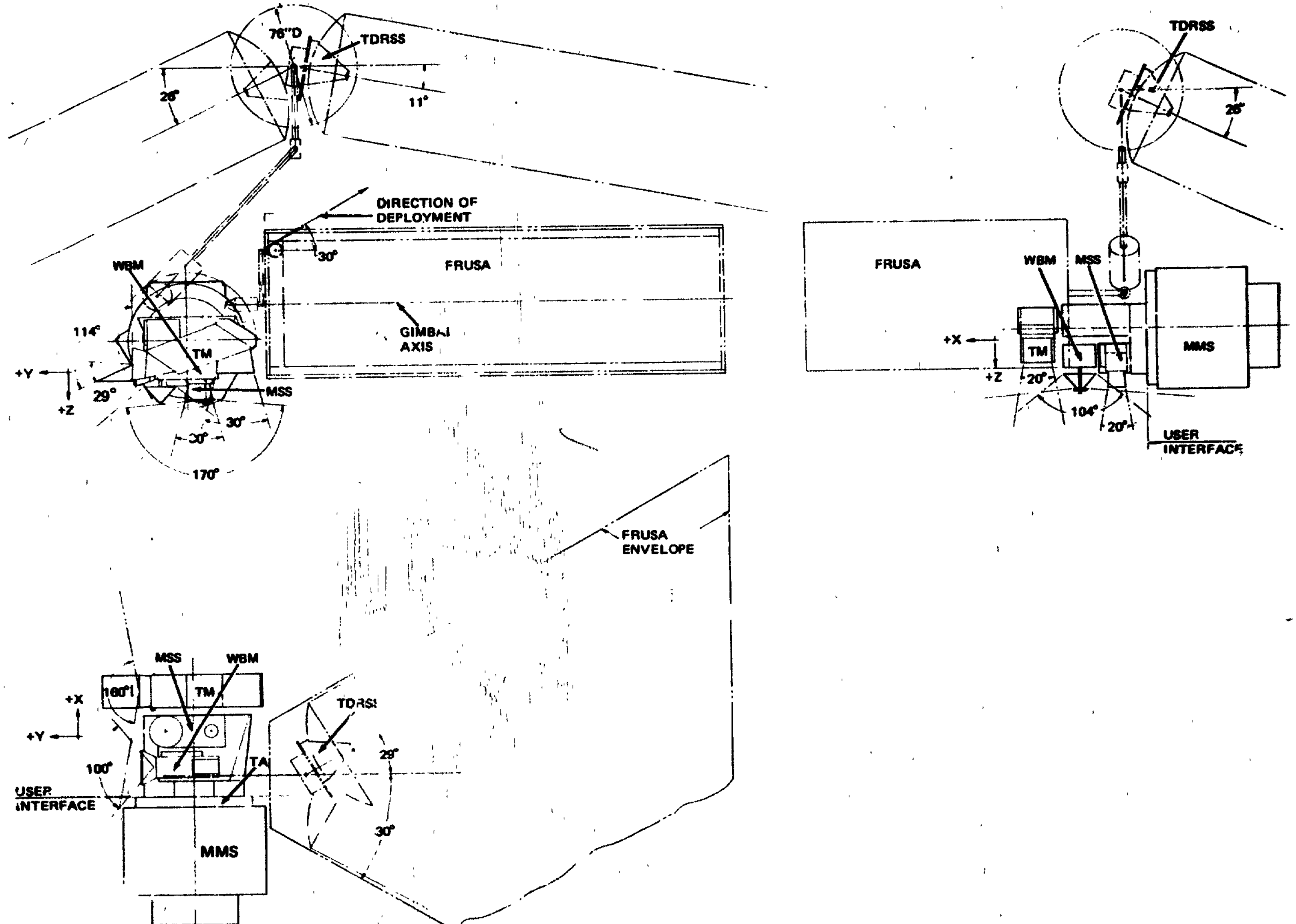


FIGURE 3.1-1 H-1A ON-ORBIT CONFIGURATION

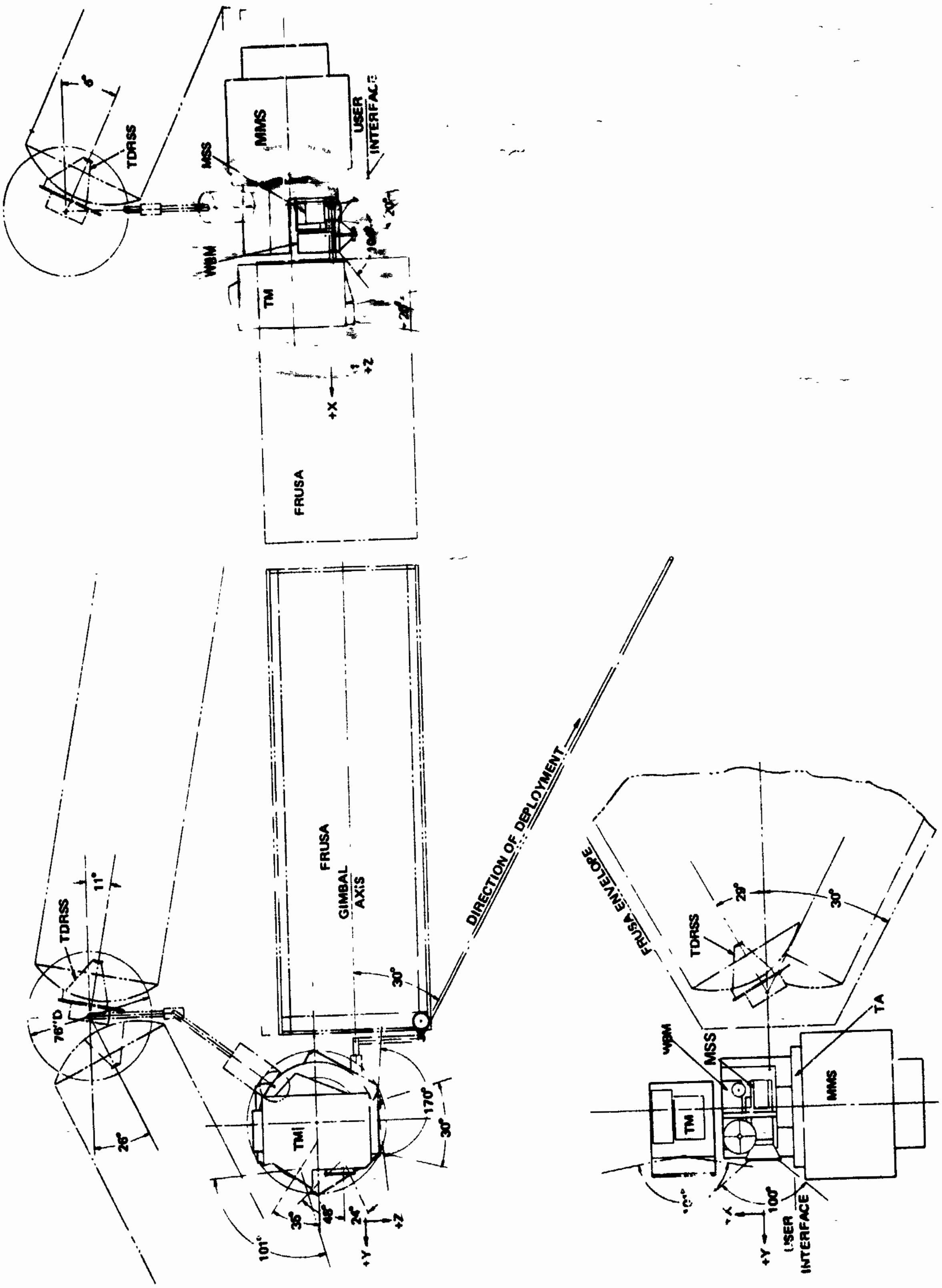


FIGURE 3.1-2 T-1A ON-ORBIT CONFIGURATION

2473-68

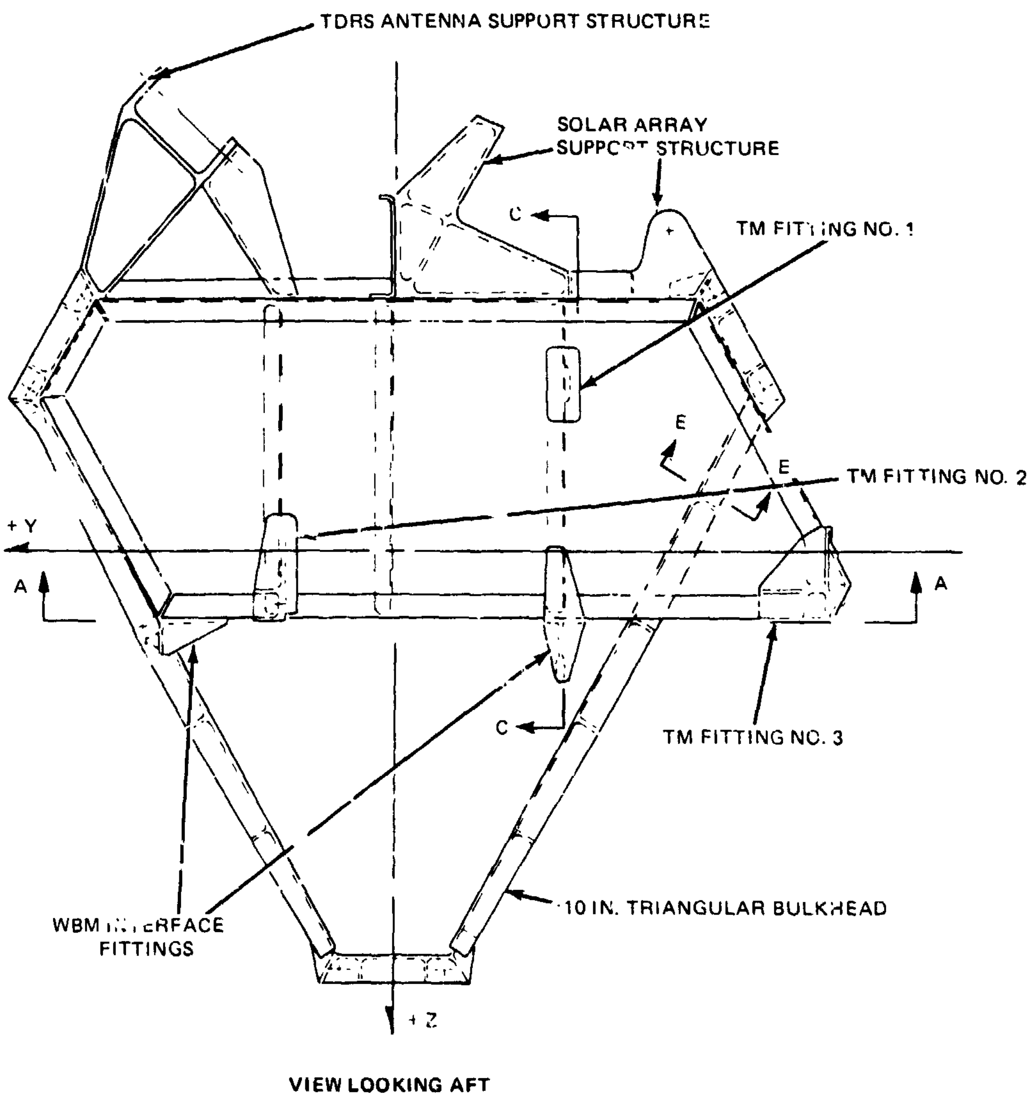
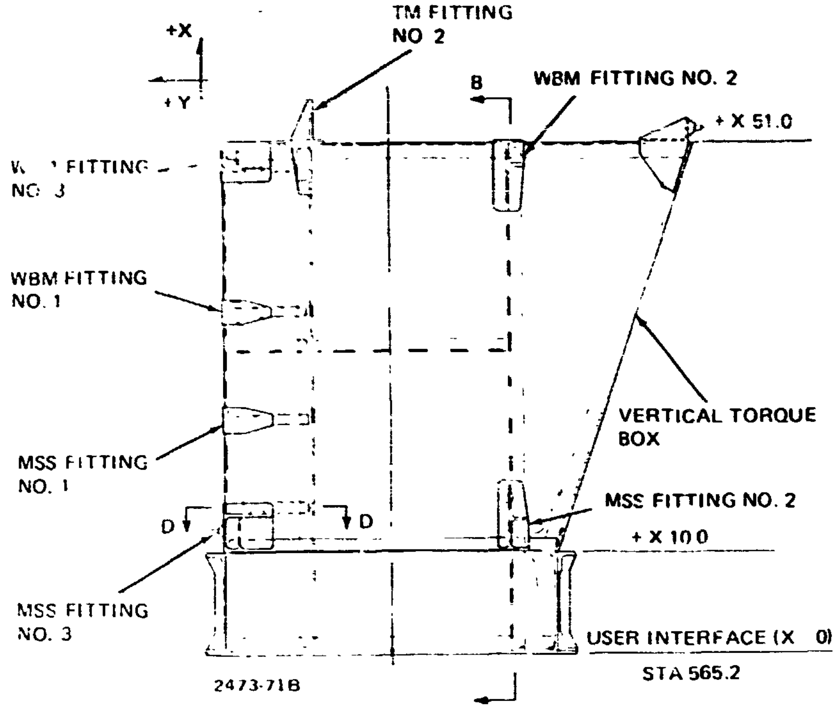
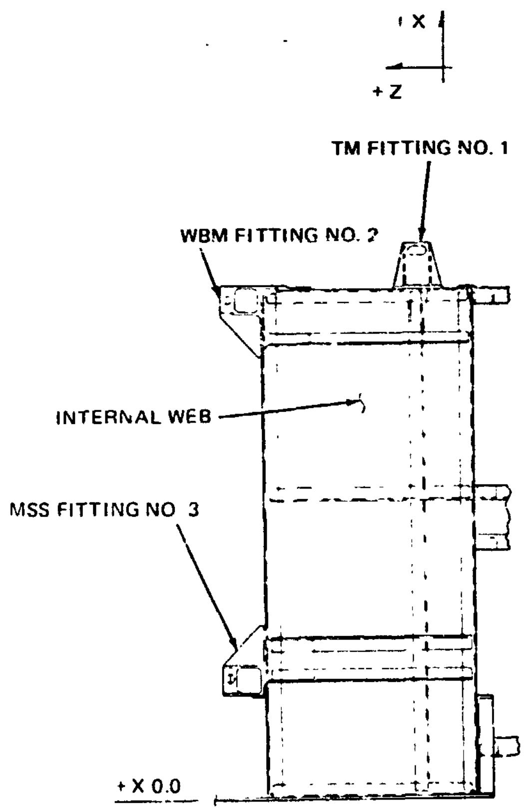


FIGURE 3.1-3A H-1A STRUCTURAL ARRANGEMENT



SECTION A-A



SECTION B-B

2473-71C

FIGURE 3.1-3B H-1A STRUCTURAL DETAILS (SHEET 1 of 5)

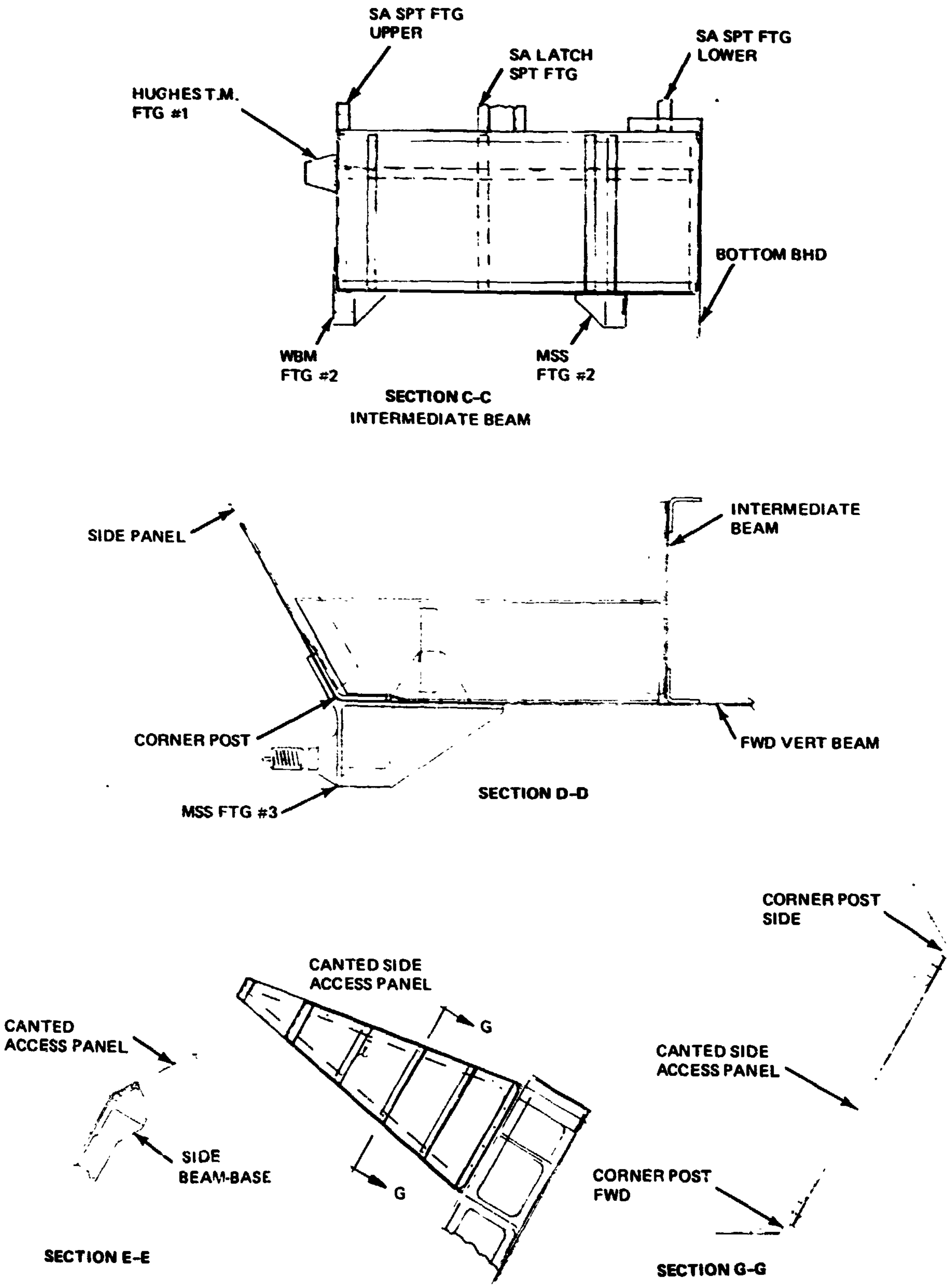
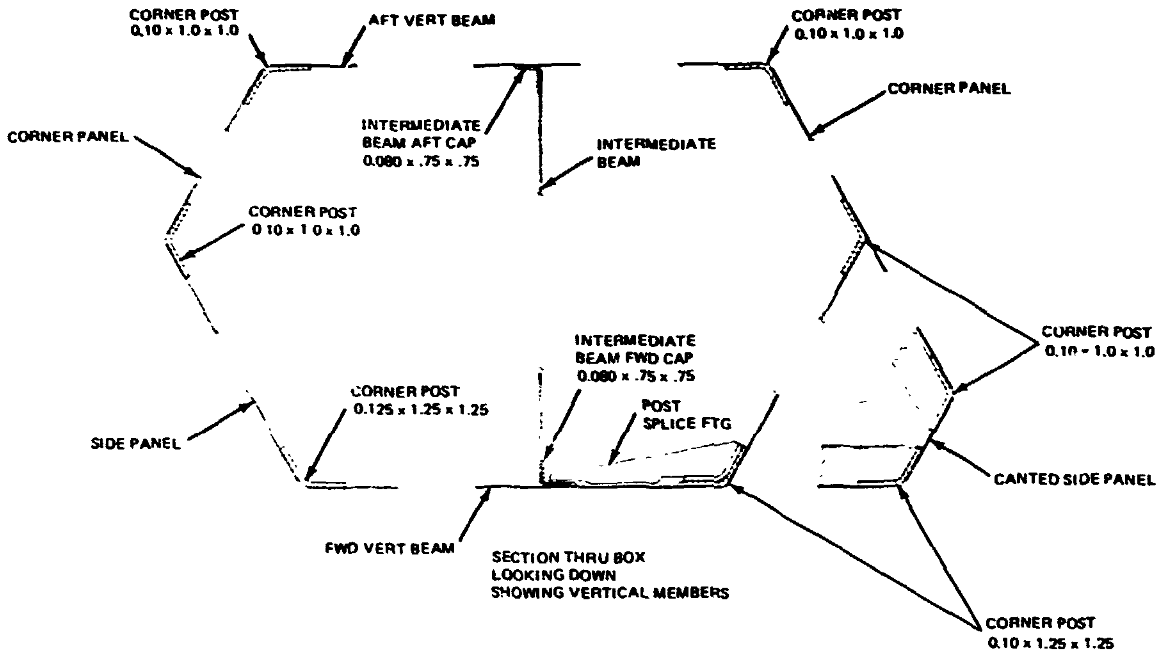


FIGURE 3.1-3B H-1A STRUCTURAL DETAILS (SHEET 2 OF 5)



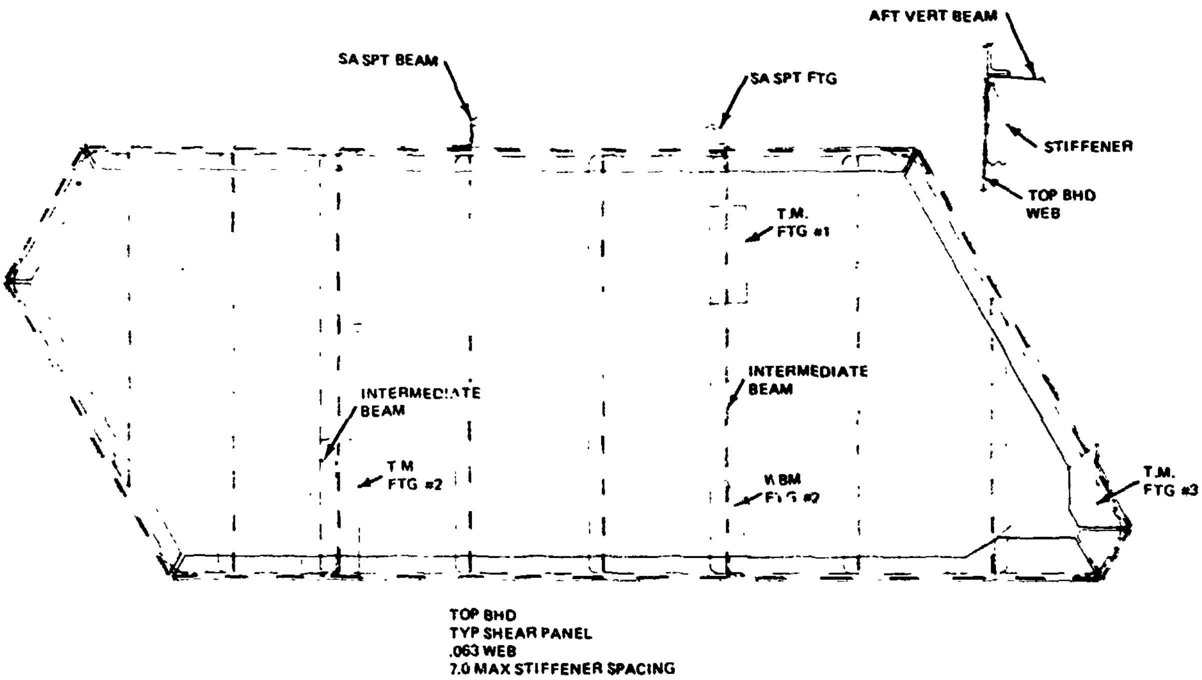
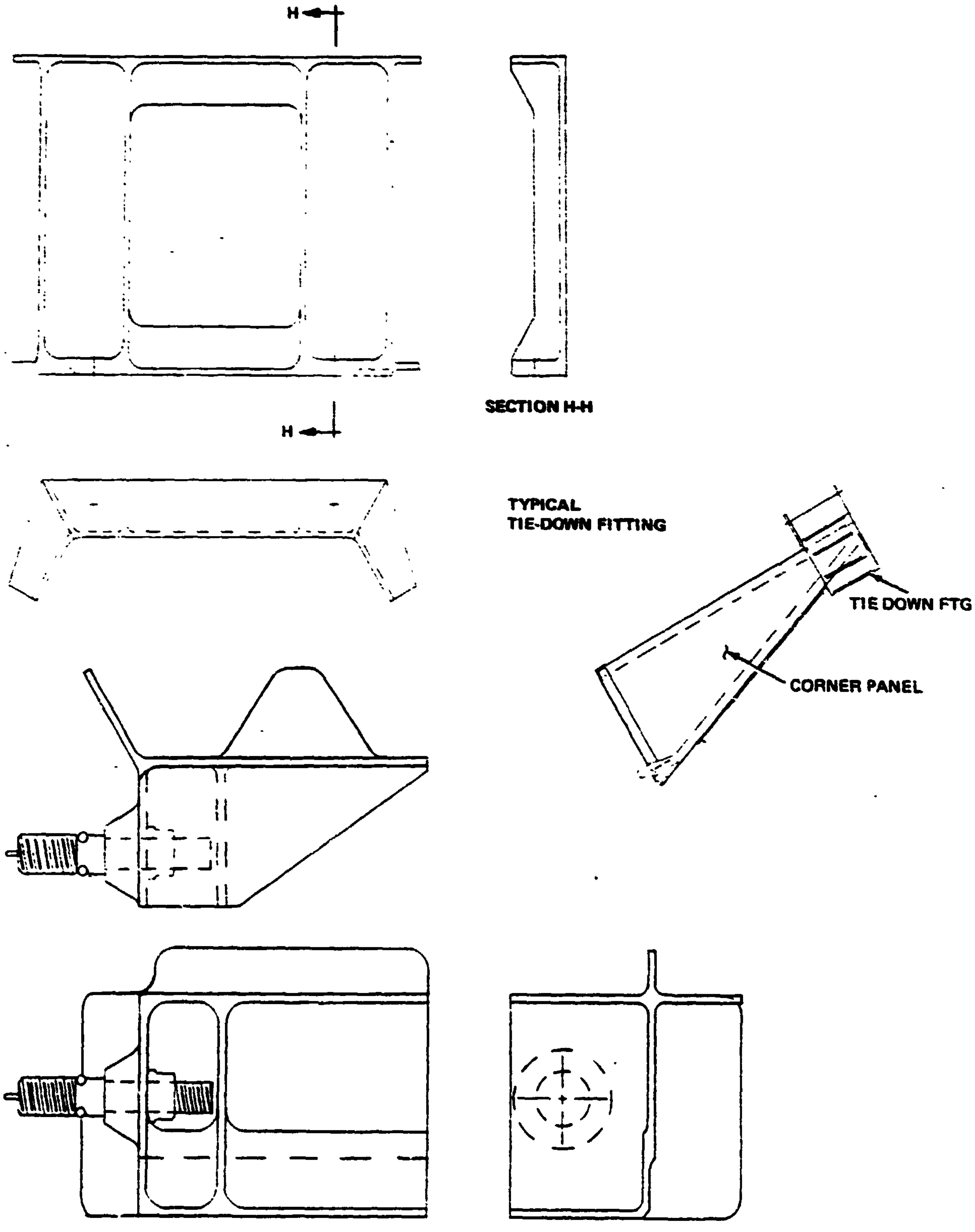
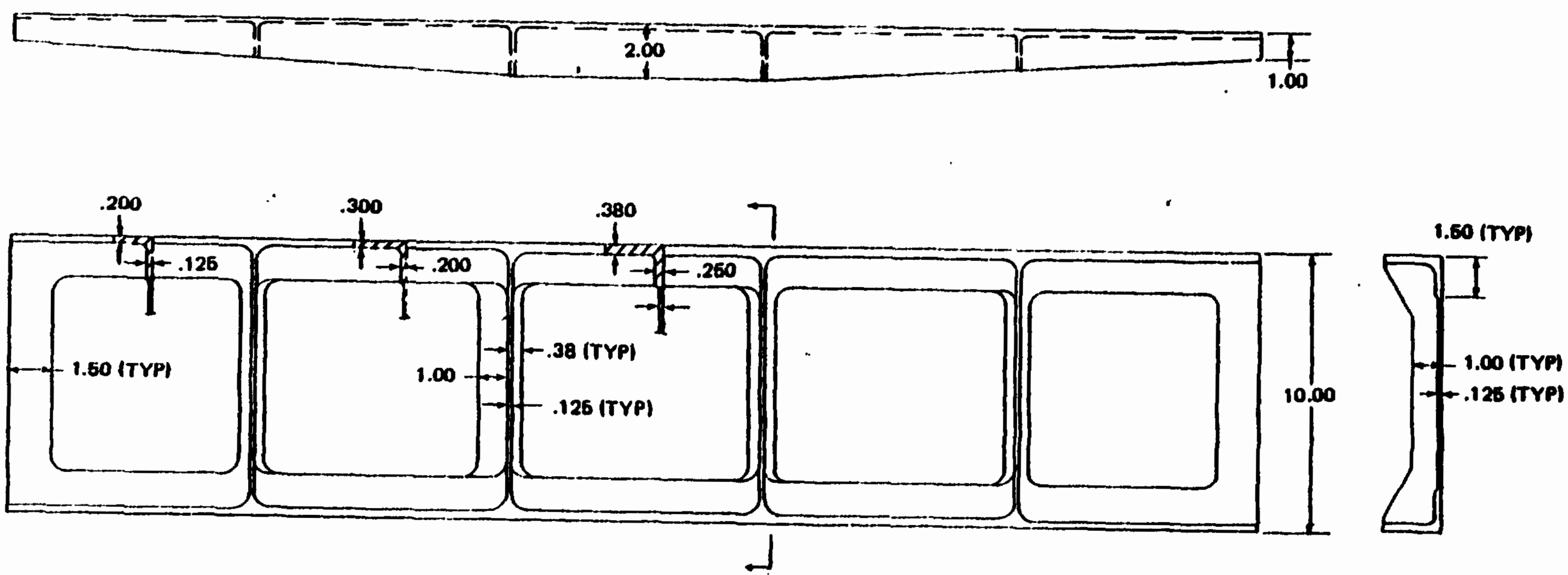


FIGURE 3.1-3B H-1A STRUCTURAL DETAILS (SHEET 3 OF 5)



MSS FITTING NO. 3

FIGURE 3.1-3B H-1A STRUCTURAL DETAILS (SHEET 4 OF 5)



TYPICAL SIDE BEAM BASE

FIGURE 3.1-3B H-1A STRUCTURAL DETAILS (SHEET 5 OF 5)

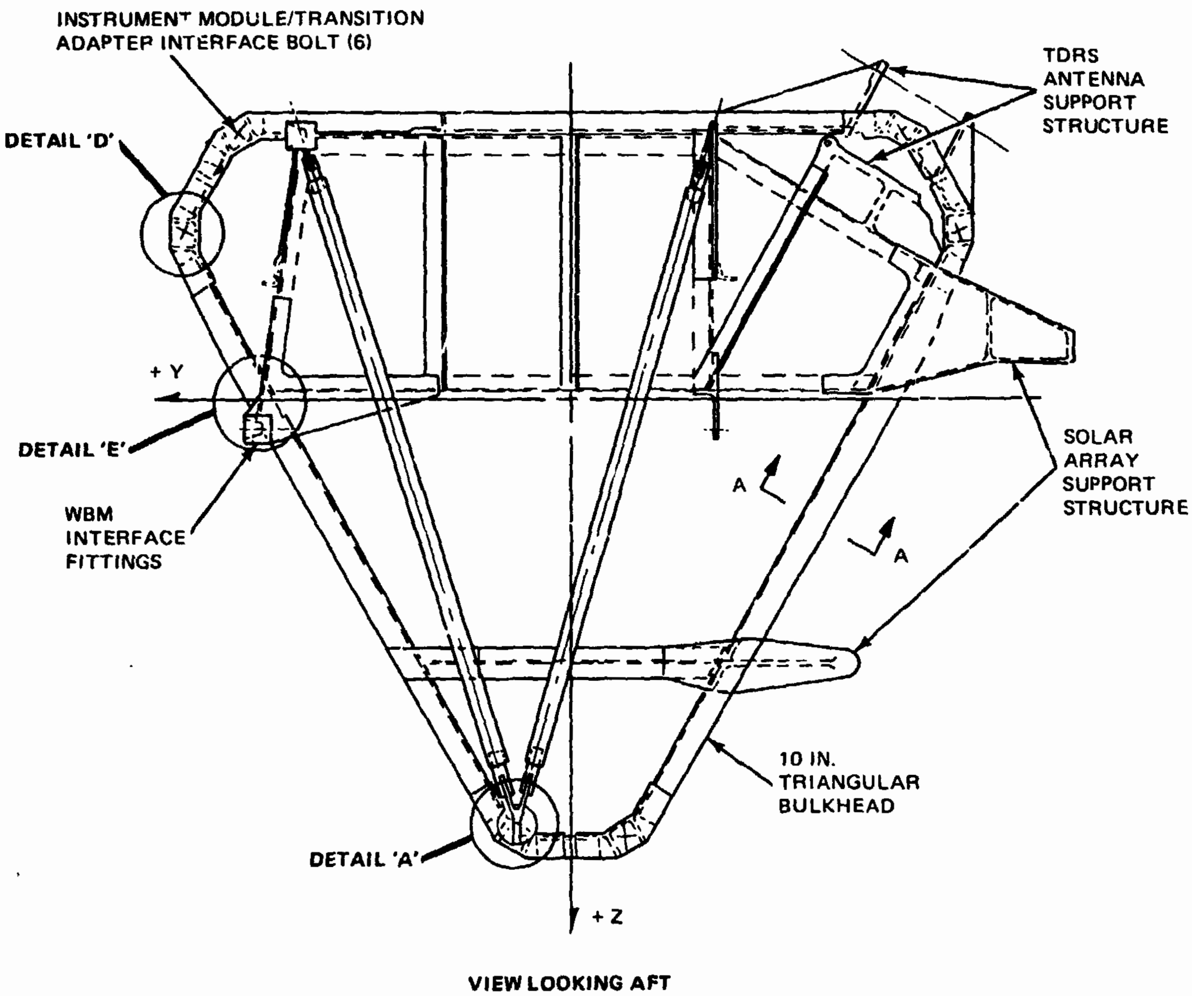
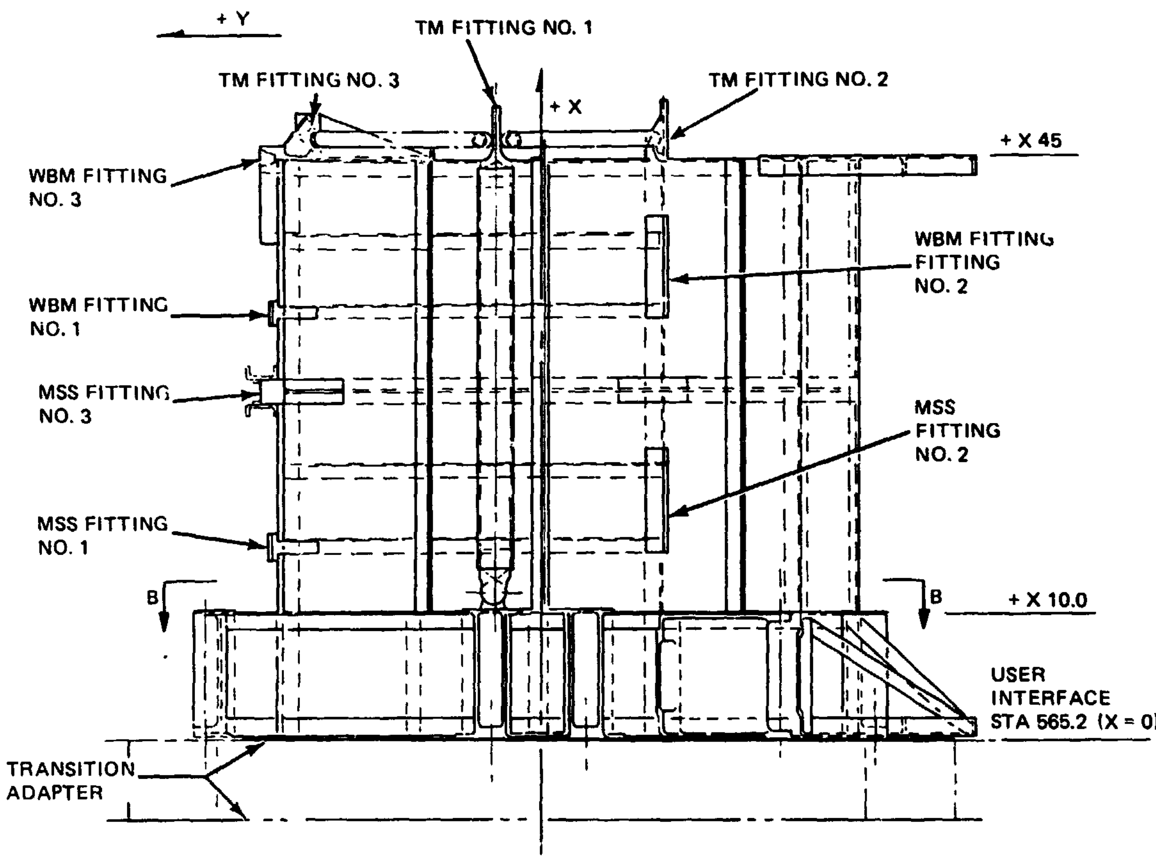


FIGURE 3.1-4A T-1A STRUCTURAL ARRANGEMENT (SHEET 1 OF3)



VIEW LOOKING INBOARD + Z SIDE

FIGURE 3.1-4A T-1A STRUCTURAL ARRANGEMENT (SHEET 2 OF 3)

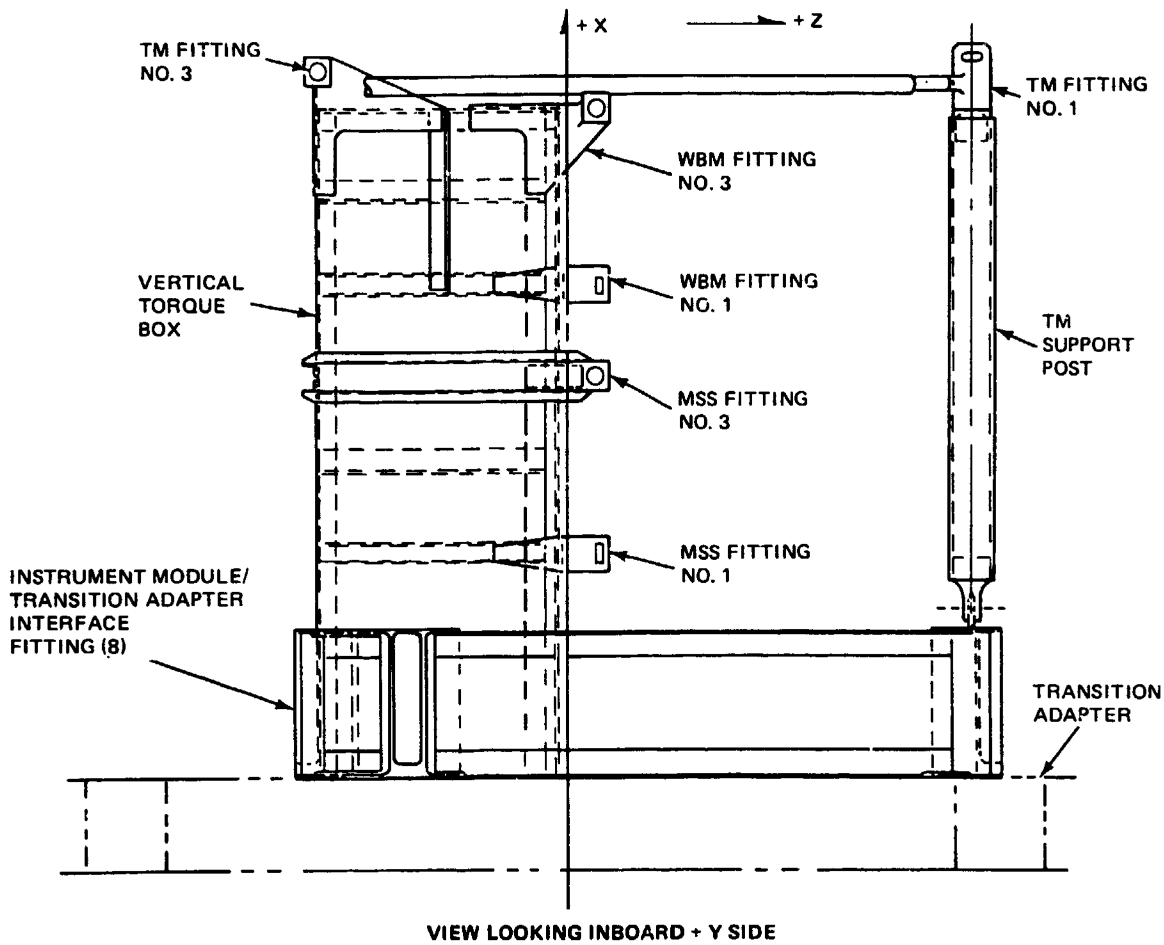


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FIGURE 3.1-4A T-1A STRUCTURAL ARRANGEMENT (SHEET 3 OF 3)

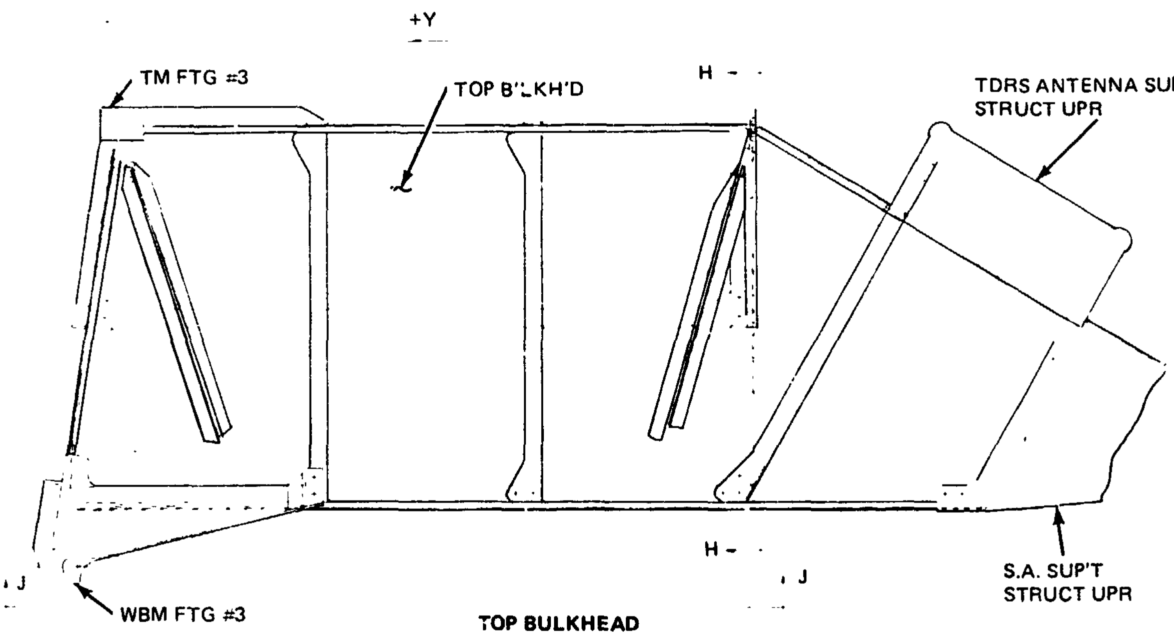
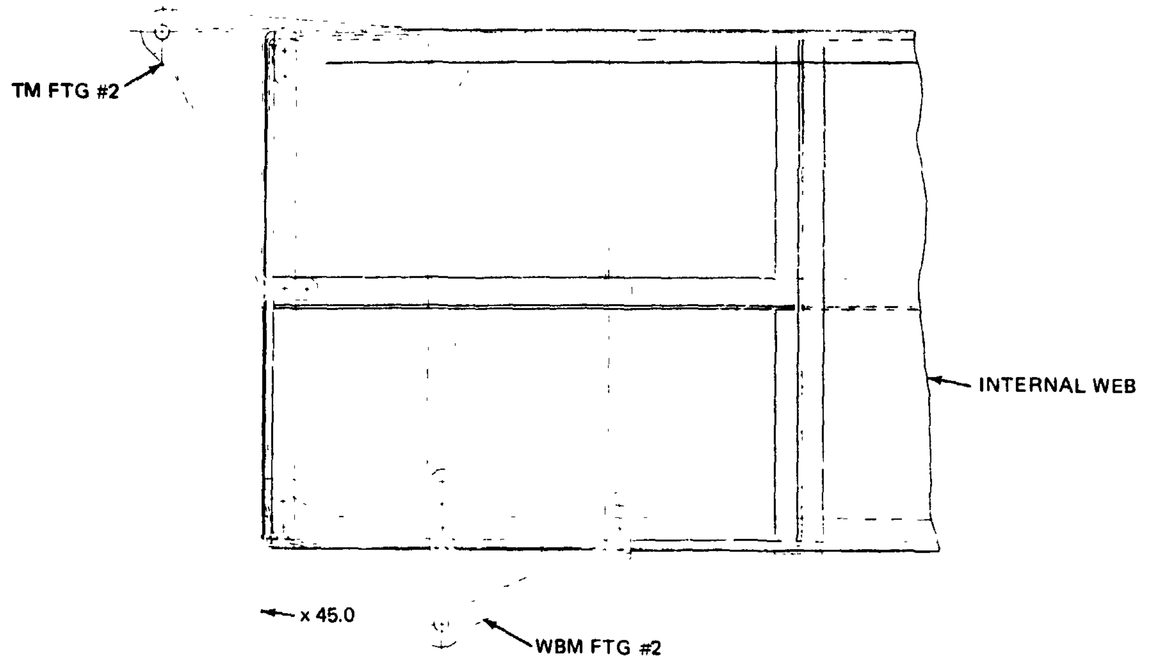
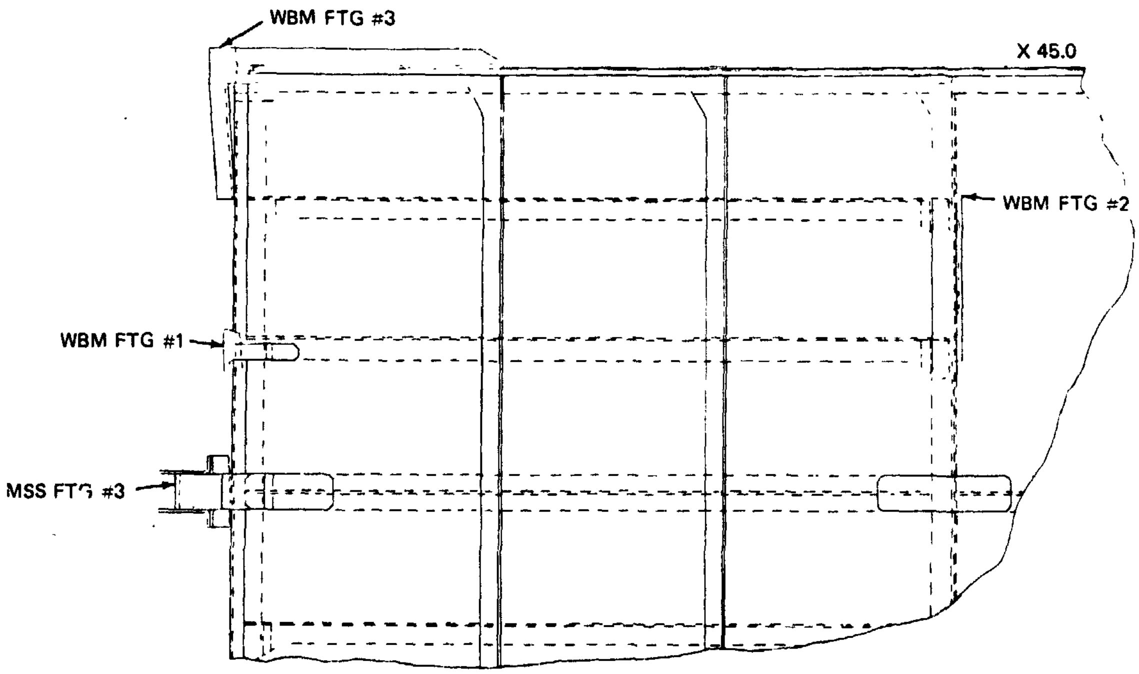


FIGURE 3.1-4B T-1A STRUCTURAL DETAILS (SHEET 1 OF 4)



SECTION H - H



SECTION J - J

FIGURE 3.1-4B T-1A STRUCTURAL DETAILS (SHEET 2 OF 4)

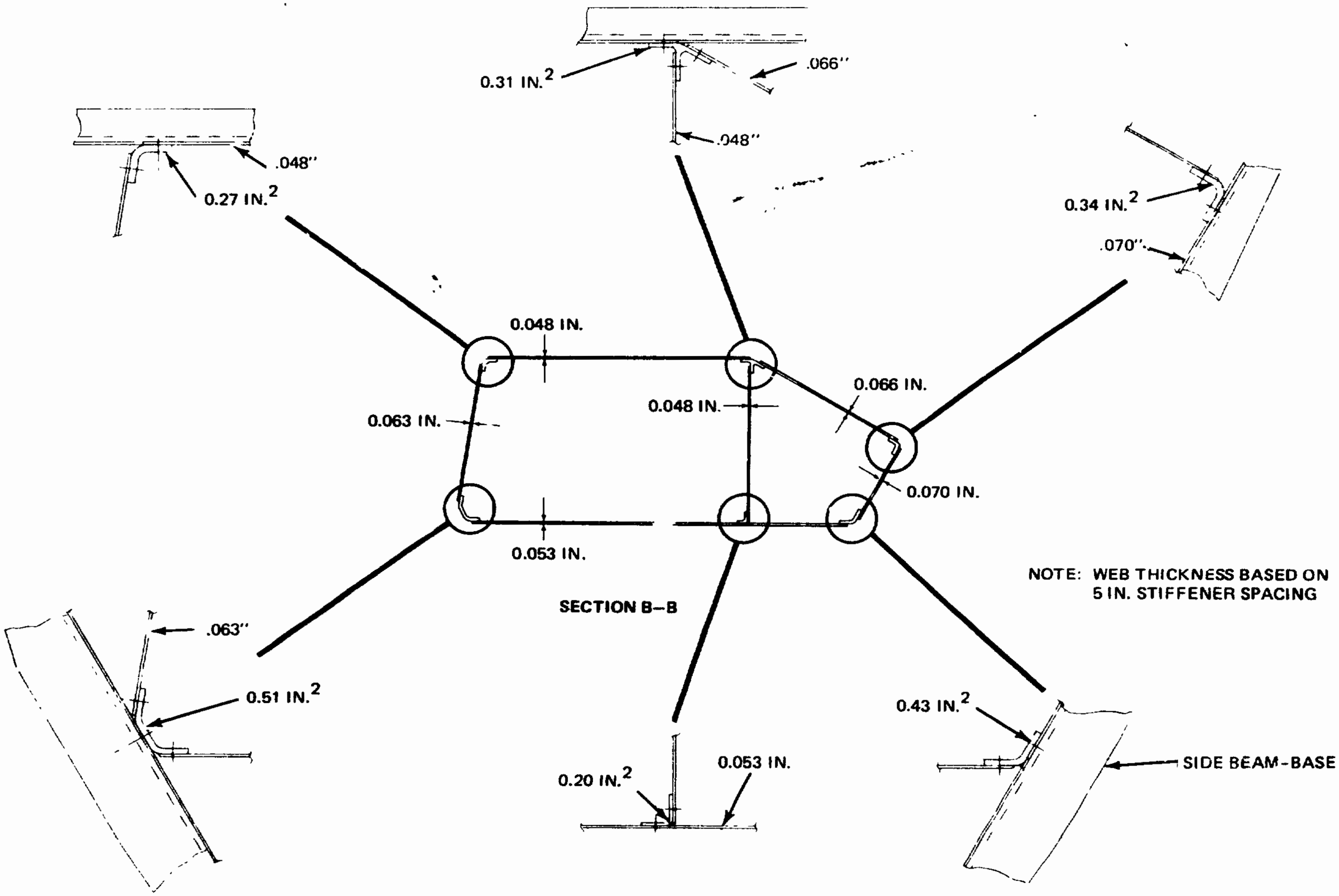


FIGURE 3.1-4B STRUCTURAL DETAILS (SHEET 3 OF 4)

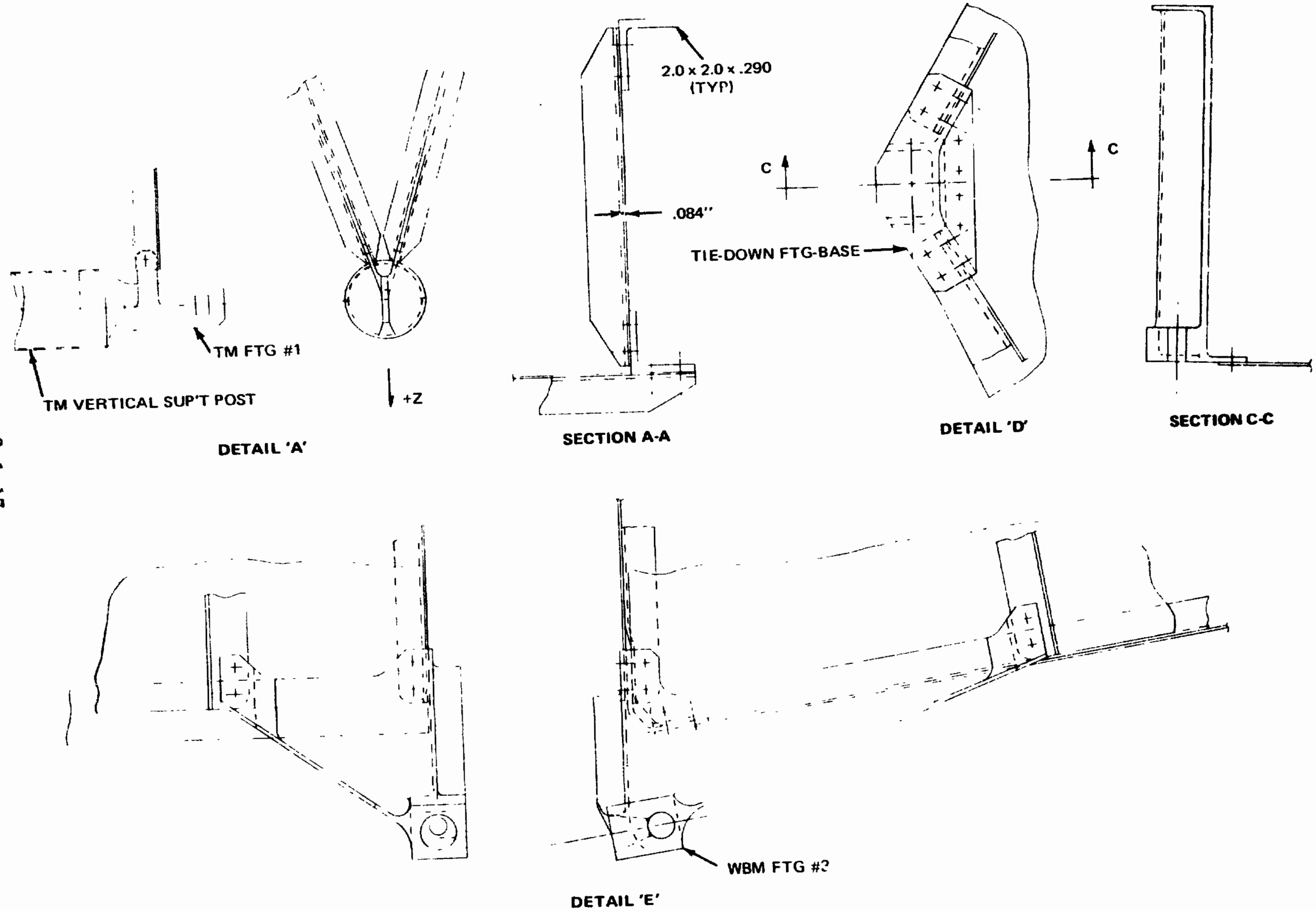
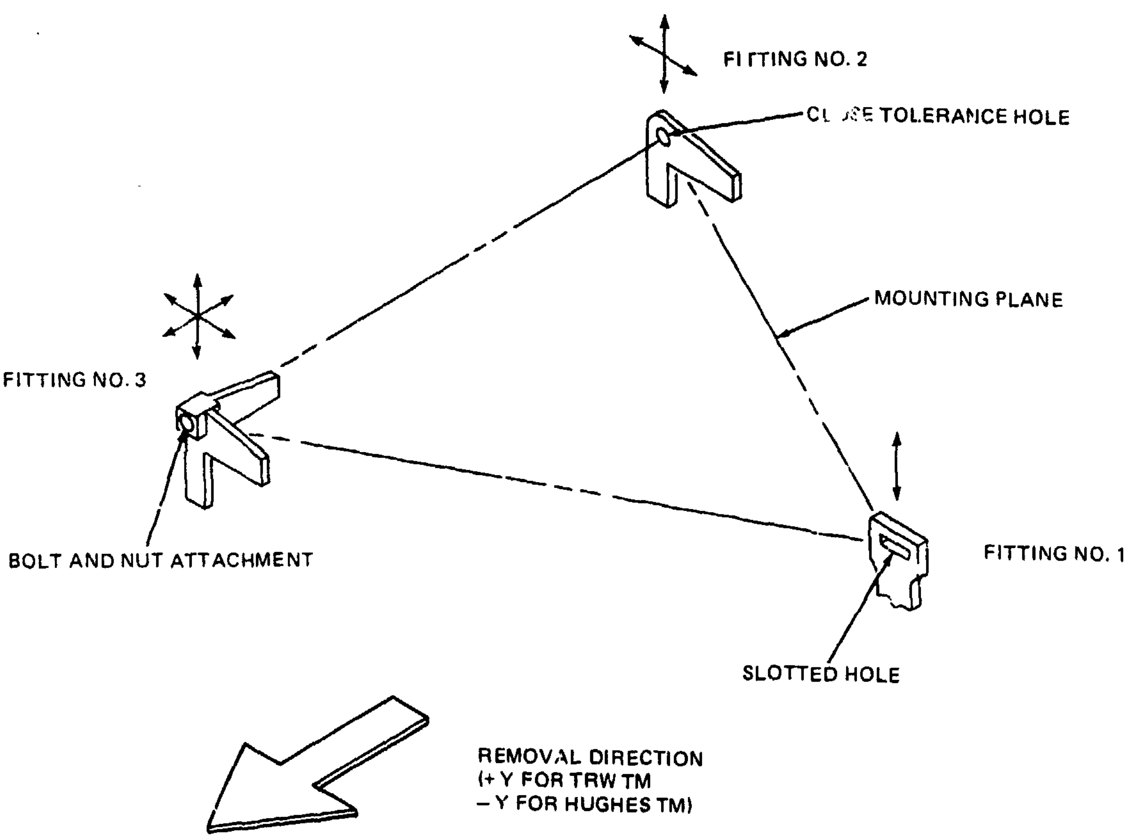


FIGURE 3.1-4B T-1A STRUCTURAL DETAILS (SHEET 4 OF 4)

The vertical torque box directly supports all of the Instrument Module components including the TDRS antenna and Solar Array appendages through discrete hard points, and redistributes the loads to the three main beams of the bulkhead. The bulkhead, in turn, beams the loads to the transition adapter.

The TM, MSS, and WBM are each attached to the primary structure through three discrete fittings. These fittings form a statically determinate mounting plane whose directional load capabilities are schematically illustrated in Figure 3.1-5. Further investigation is required to determine if the proposed load directions are compatible with the internal structure of the various mounted components. Principal coordinates, attachment points, and CG locations for the T-1A and H-1A configurations are shown in Figures 3.1-6 and 3.1-7 respectively.

The structure is aluminum alloy throughout, except for the instrument interface fittings which are titanium (6A1-4V). Sheet aluminum is 2024-T81. Machined parts and extrusions are manufactured from 7075-T73 aluminum alloy to minimize the possibility of stress corrosion.



2473-147

FIGURE 3.1-5 TM INTERFACE LOAD DEFINITION

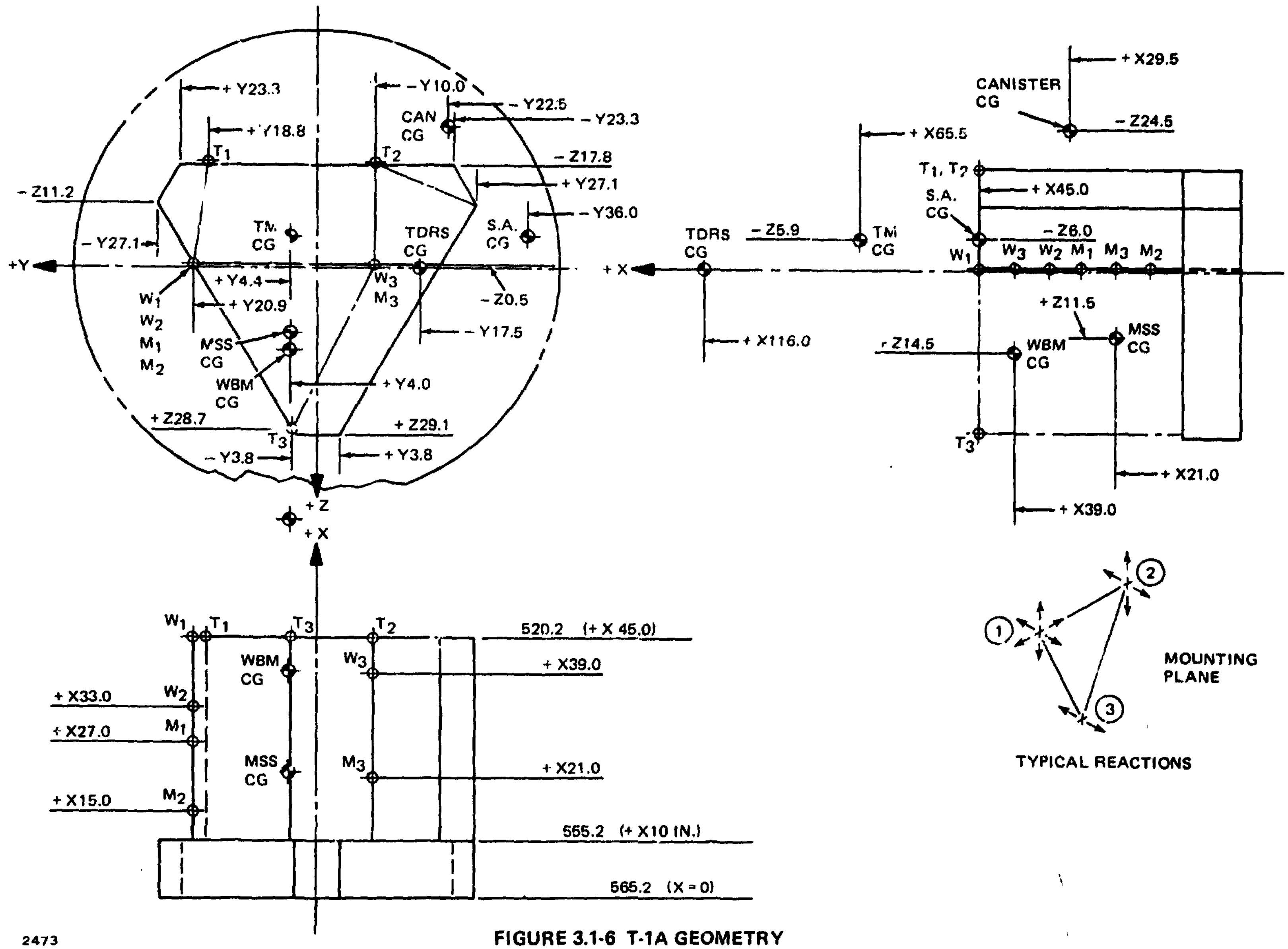


FIGURE 3.1-6 T-1A GEOMETRY

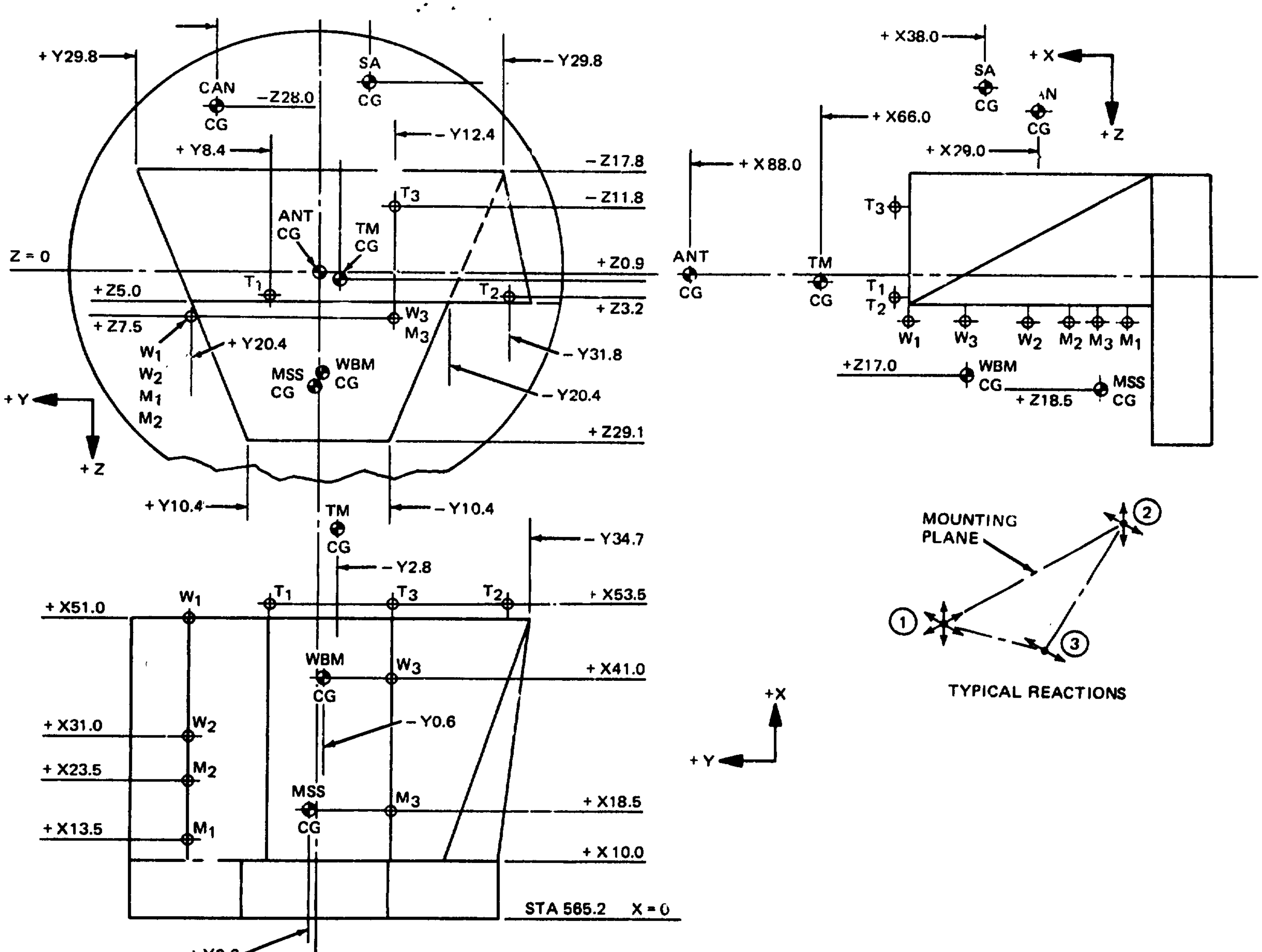


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FIGURE 3.1-7 H-1A GEOMETRY

3.2 MECHANICAL SYSTEMS

Three areas of mechanical design were investigated in some detail during the Phase II Concept Validation Study. These investigations included:

- A design for the Solar Array configuration and its deployment
- A design for deployment of the TDRS antenna
- A conceptual design for the mounting of equipment that will be compatible with the Module Exchange Mechanism (MEM) for on-orbit resupply.

Each design investigation is discussed below, providing a statement of the requirements, the system constraints, the design approach used, and a brief summary of the pertinent features associated with the recommended design.

3.2.1 Solar Array

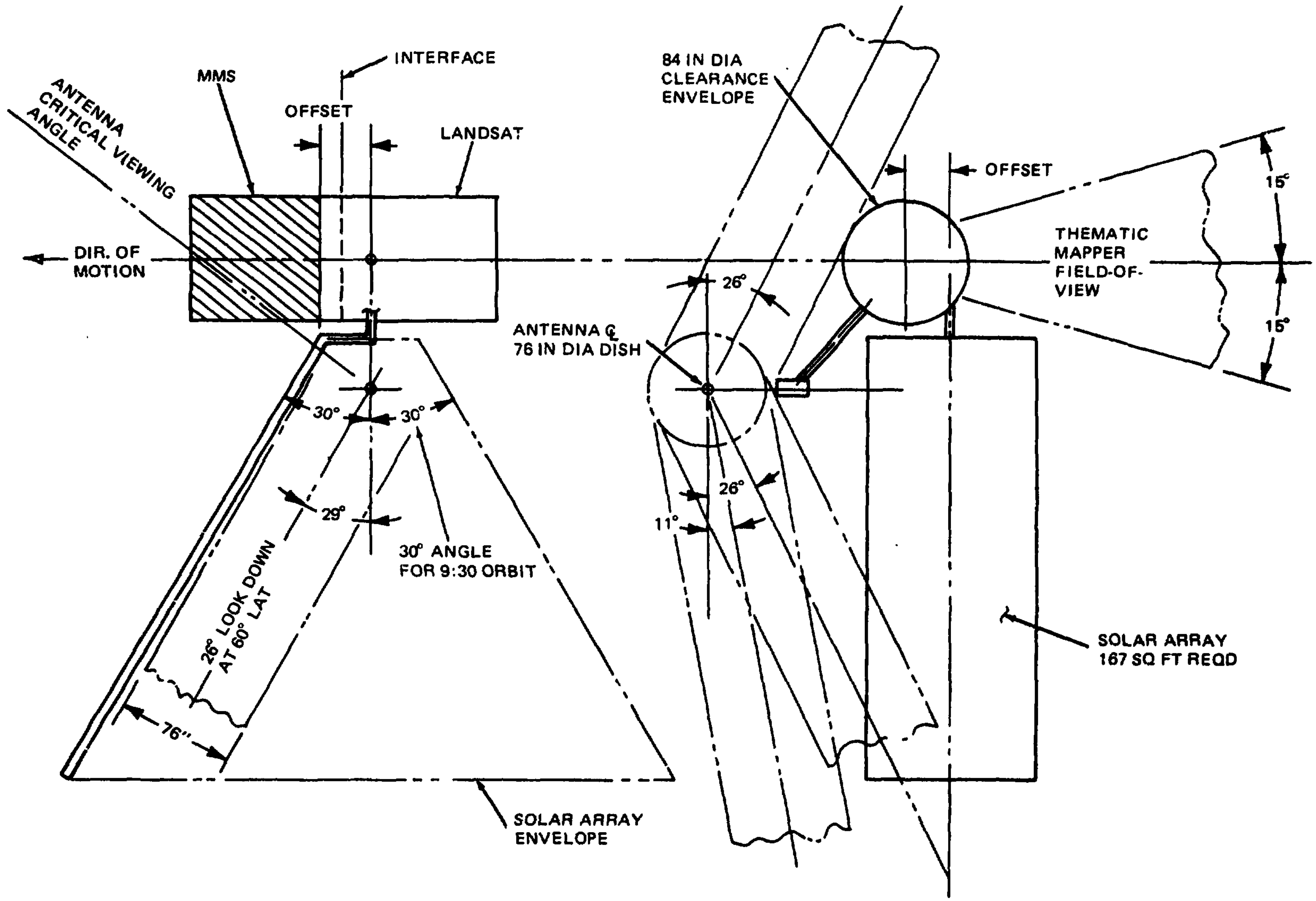
In the investigation of the Solar Array configuration for the Landsat follow-on satellite, it was found that a major consideration was the look-down angle of 25.8 deg required for the TDRS antenna. While the approach of clearing the critical look-down angle will insure that no interference ever takes place, it places large penalty on the design of both the Solar Array and the TDRS antenna mast height.

A preliminary investigation of the interrelation between antenna position and Solar Array shape was made using the following assumptions:

- Landsat satellite at 705 km, 0930 sun-synchronous orbit with inclination angle of 90° rather than the actual 98° -14'
- Mapping takes place only over portions of the earth's surface in daylight
- Solar array power nominal 1500 watts
- Two geostationary satellites will receive all the signals from the 76-in. diameter antenna. These satellites are located at longitude 45°W and longitude 168°W at latitude 0°.

3.2.1.1 General Configuration

The results of this preliminary investigation have indicated that a Solar Array geometry of the type shown in Figure 3.2-1 is most suitable and that the height of the antenna above the centerline of Landsat will be controlled by clearance over the



2473-128

FIGURE 3.2-1 SOLAR ARRAY/ANTENNA GEOMETRY AND SCHEMATIC ARRANGEMENT

forward end of the actual vehicle rather than by the Solar Array. It should be noted that the use of either the existing FRUSA or a modified version appears to present no obscuration problems, eliminating the need and cost associated with designing and qualifying a new configuration of Solar Array. Figures 3.2-2 through 3.2-8 show the Solar Array and antenna general geometry and lines-of-sight for a series of satellite latitude locations ranging from 15° to 90°.

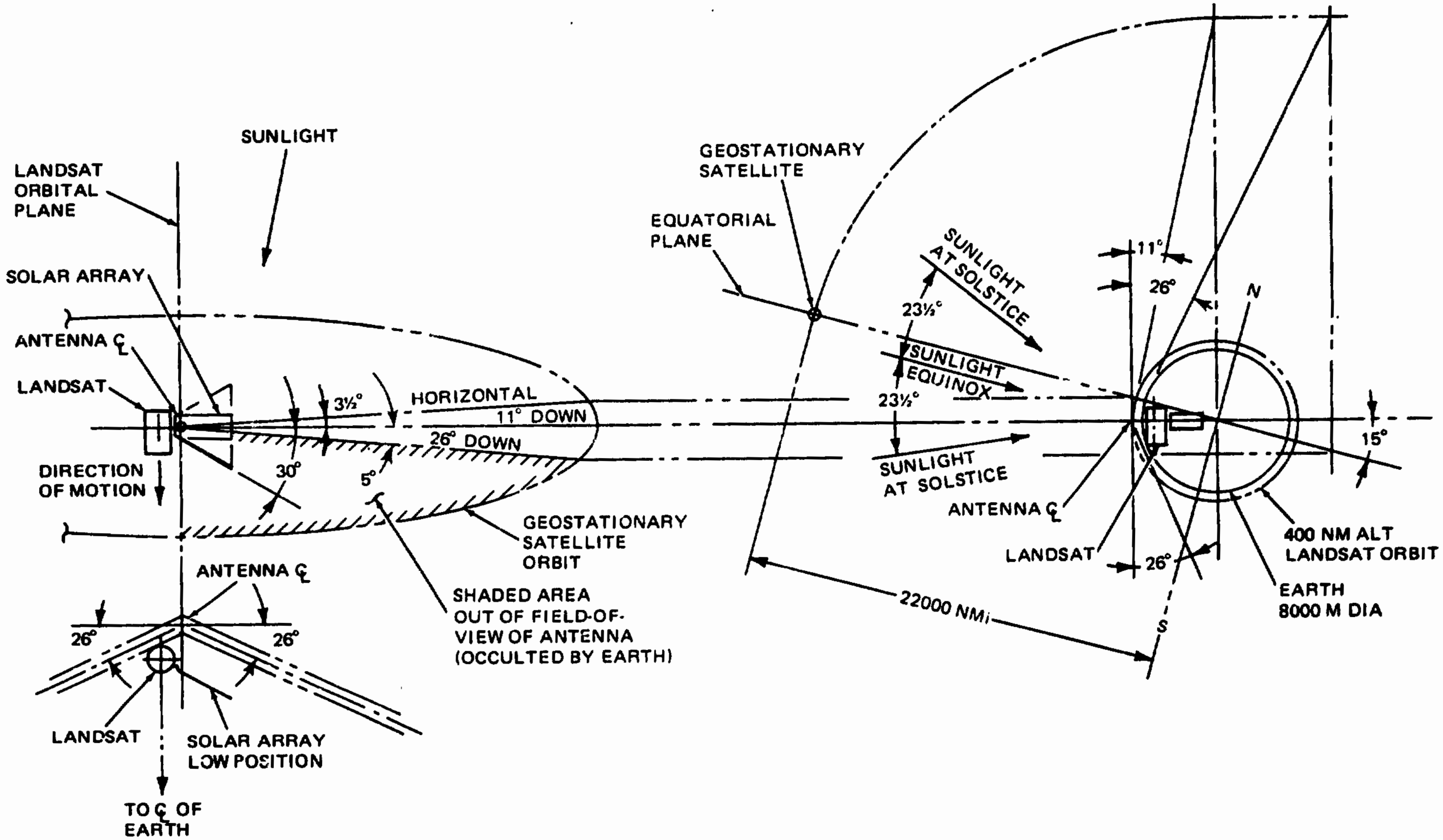
An interesting but incidental item uncovered in this investigation is the fact that a portion of the surface of the Earth is always out of the field-of-view of the two geostationary satellites operating with Landsat at an orbital altitude of 705 km. This blind area encompasses portions of Afghanistan, Pakistan, Nepal and most of India.

3.2.1.2 Solar Array for H-1A and T-1A Configuration

Figure 3.2-1 defines a Solar Array and antenna system compatible with orbital requirements but essentially independent of an actual vehicle configuration. Refer to Figures 2.6-2 through 2.6-5 for three possible configurations of the Solar Array for the Hughes TM configuration and one arrangement for the Instrument Module incorporating the TRW TM. In all of these configurations, the Solar Array has been shown as a FRUSA since it packages so well and is a proven concept. Figure 3.2-9 shows the design details of the Solar Array for the H-1A configuration. No details of the Solar Array design for the T-1A configuration were prepared since it is essentially the same as that for H-1A. The only differences are the mounting location and the amount of angular rotation required for deployment.

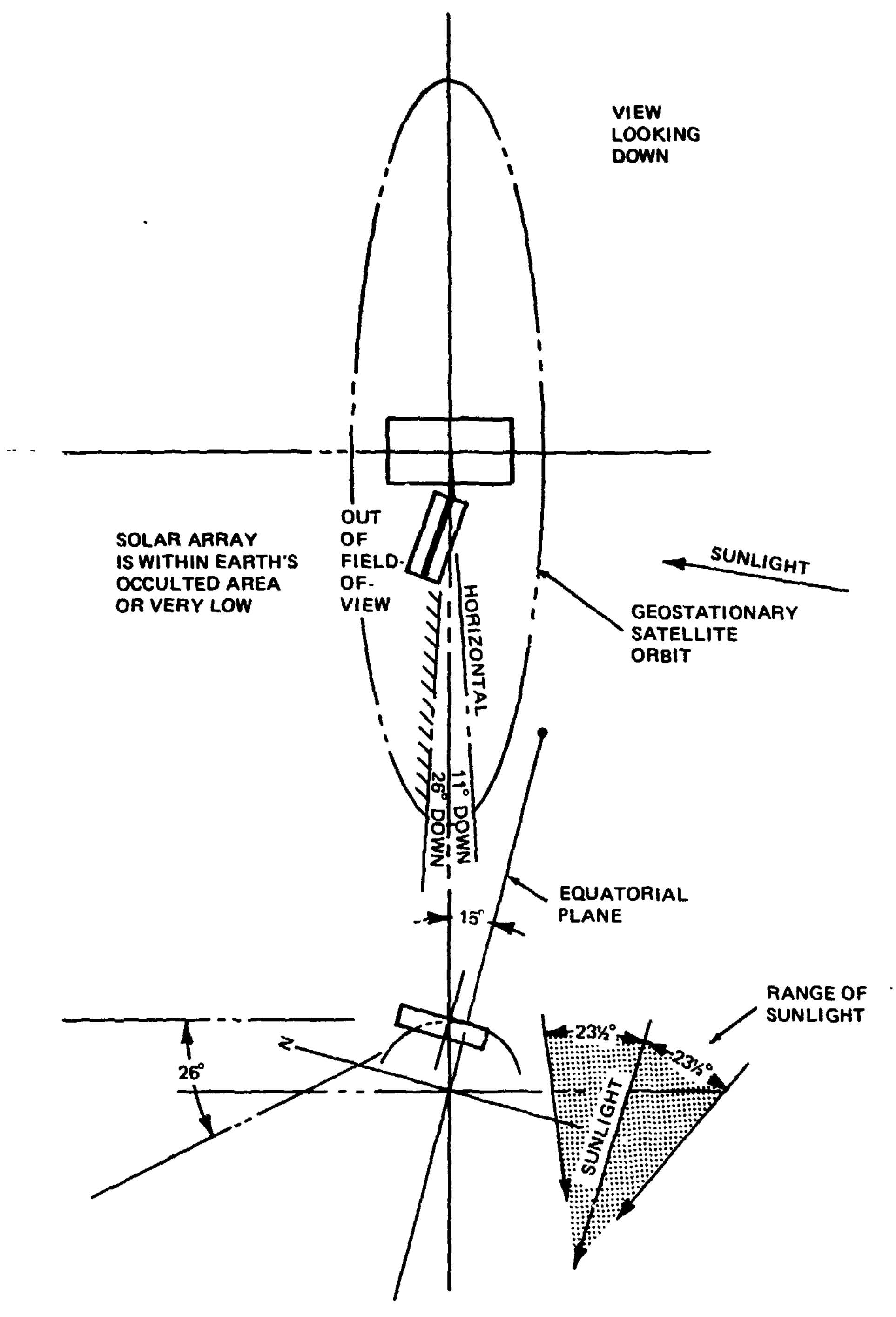
3.2.1.3 Solar Array Operation

A brief description of the equipment components and their mode of operation follows. In the launch position, all of the Solar Array subsystem components are within the 84-inch diameter clearance circle of the Delta shroud. The Solar Array itself is a modified FRUSA, 25 feet long and 6.7 feet wide, rolled up on a drum to withstand the launch environment. The FRUSA drum is enclosed in a protective canister which is in turn preloaded against two stop brackets by a locking hook mechanism.



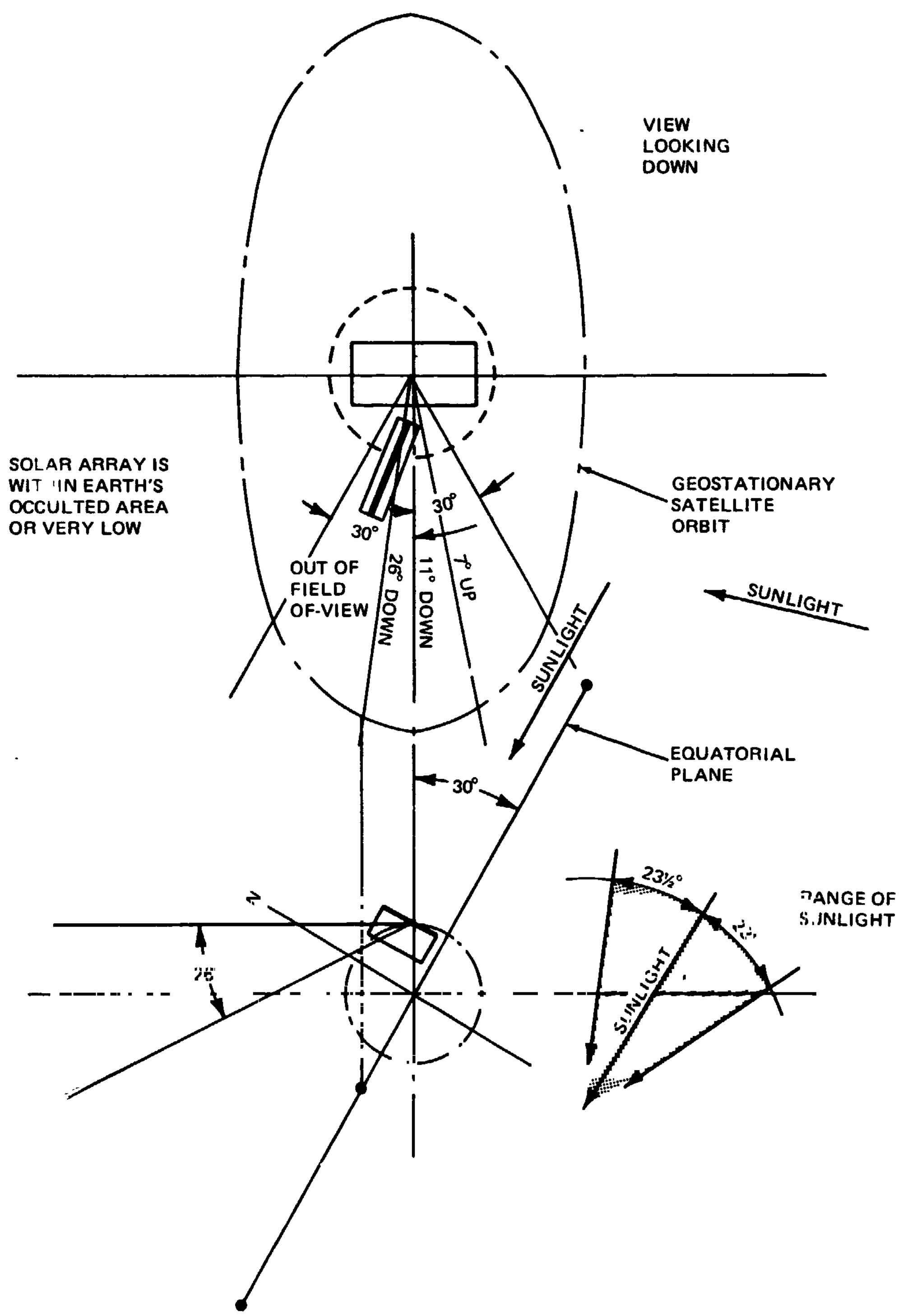
2473-129

FIGURE 3.2-2 SOLAR ARRAY/ANTENNA GENERAL GEOMETRY AT LATITUDE 15°



2473-129

FIGURE 3.2-3 SOLAR ARRAY/ANTENNA LINES-OF-SIGHT, LATITUDE 15°



2473-130

FIGURE 3.2-4 SOLAR ARRAY/ANTENNA, LINES-OF-SIGHT, LATITUDE 30°

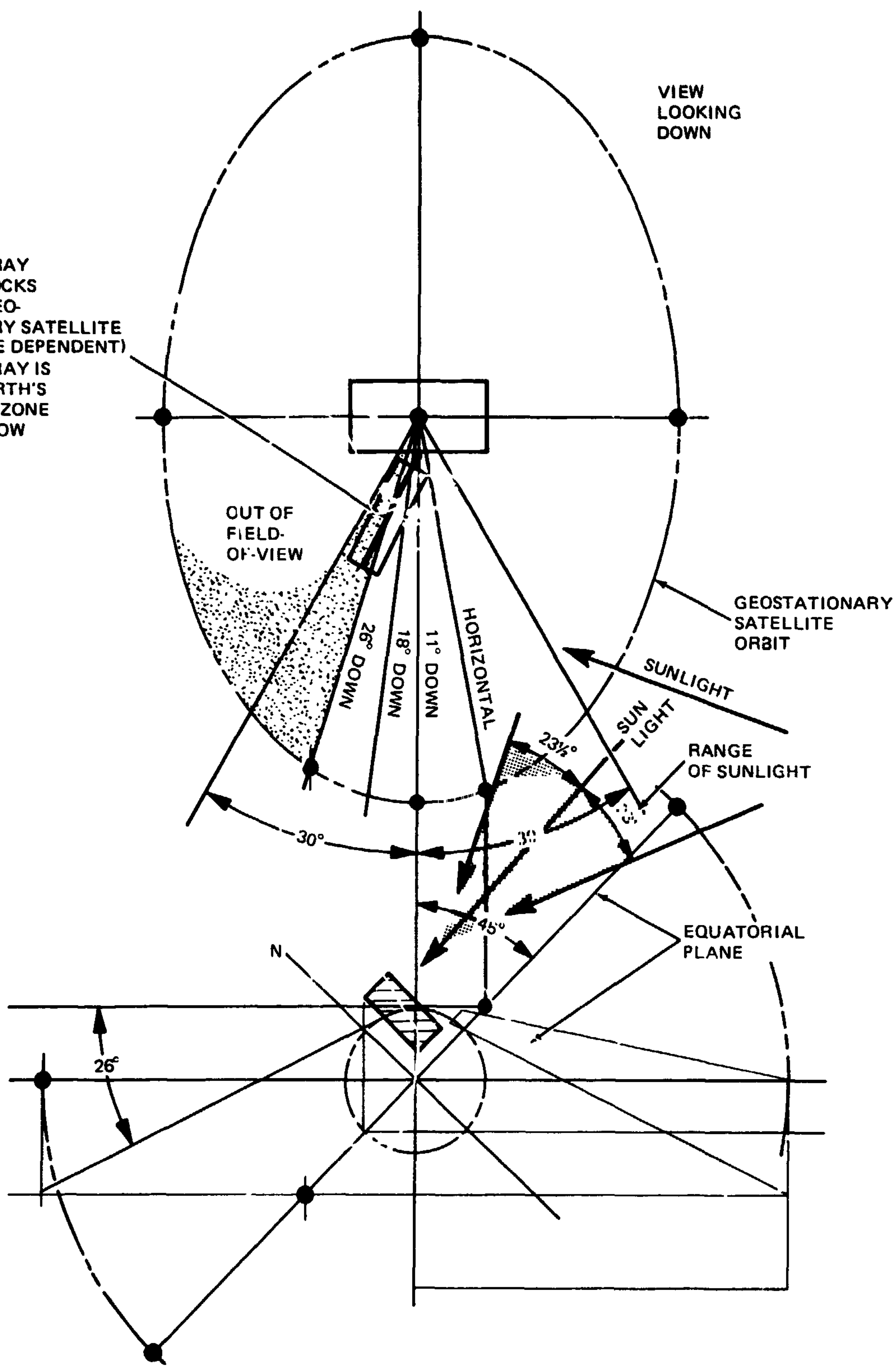
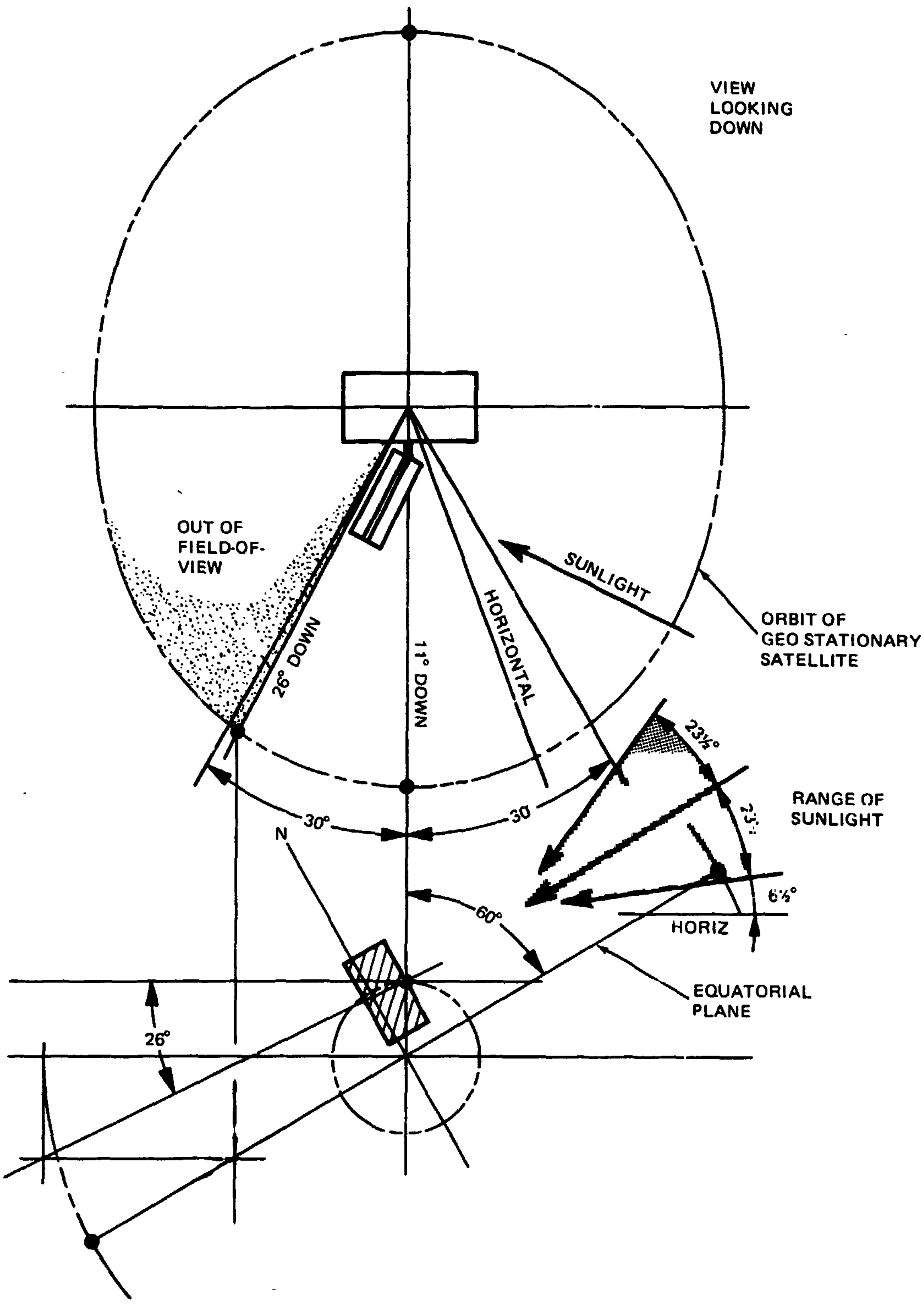


FIGURE 3.2-5 SOLAR ARRAY/ANTENNA, LINES-OF-SIGHT, LATITUDE 45°



2473-132

FIGURE 3.2-6 SOLAR ARRAY/ANTENNA, LINES-OF-SIGHT, LATITUDE 60°

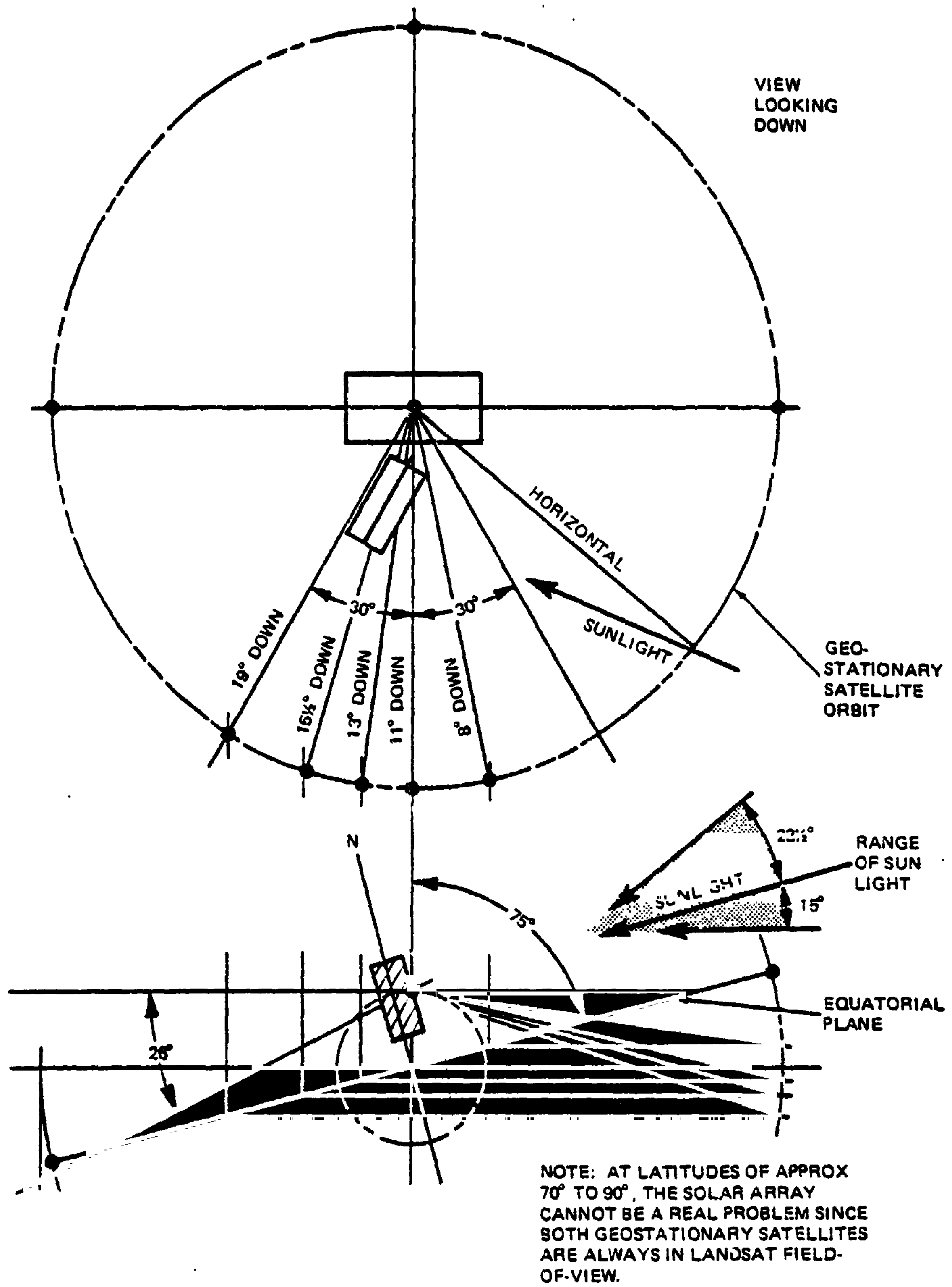
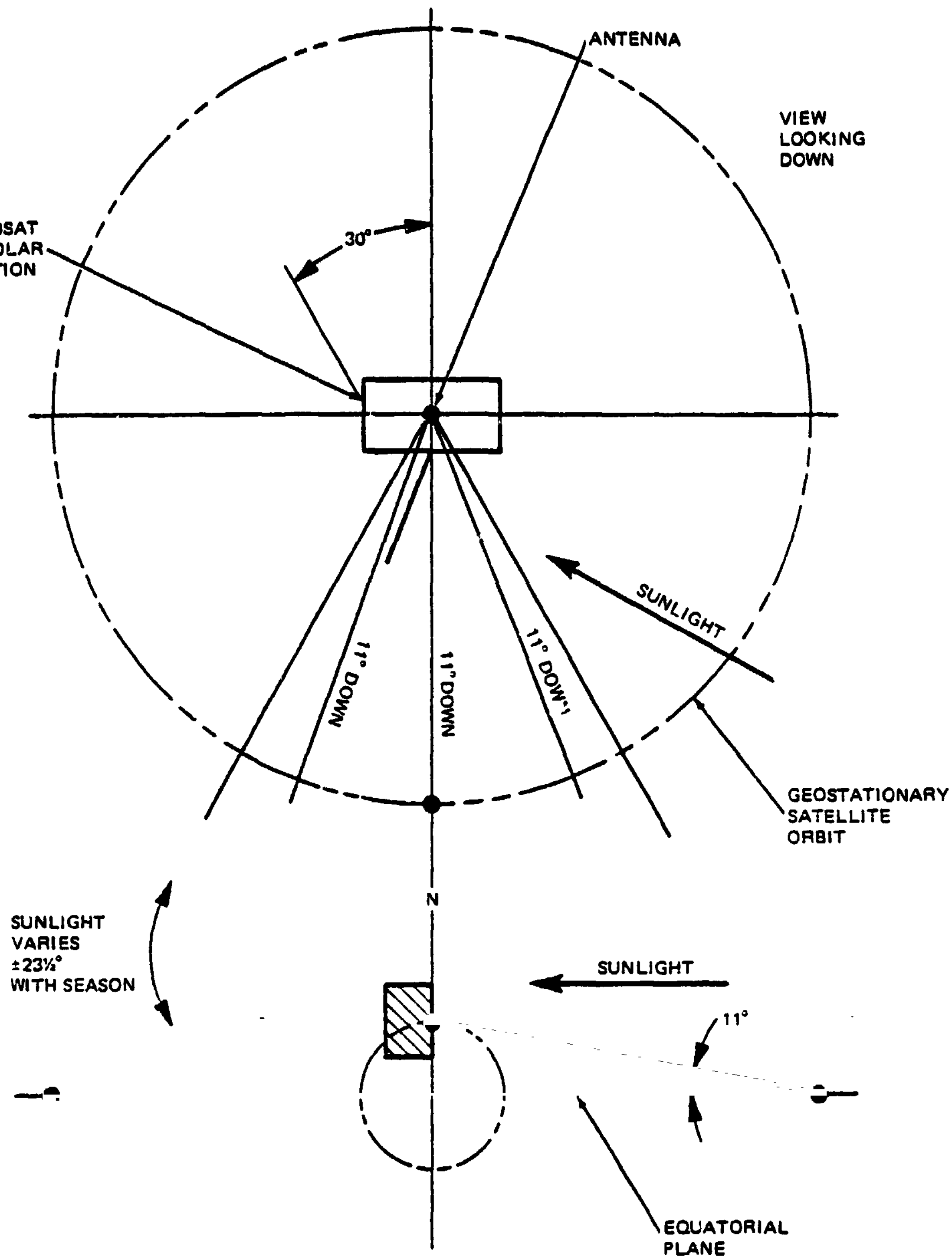


FIGURE 3.2-7 SOLAR ARRAY/ANTENNA, LINES-OF-SIGHT, LATITUDE 75°

2473-133



2473-134

FIGURE 3.2-8 SOLAR ARRAY/ANTENNA, LINES-OF-SIGHT LATITUDE 90°

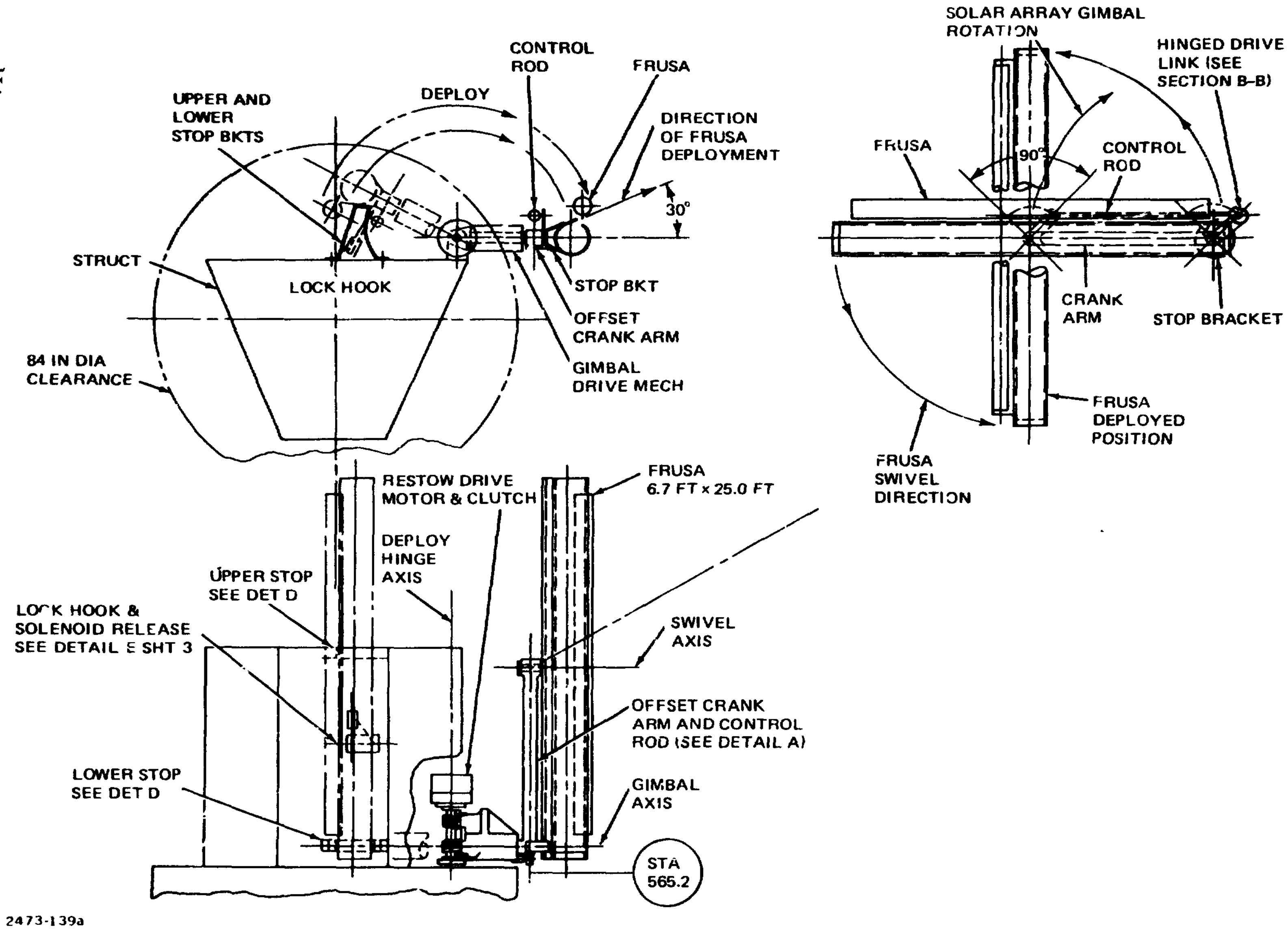


FIGURE 3.2-9 LANDSAT H-1A SOLAR ARRAY ASSEMBLY AND DETAILS (SHEET 1 OF 3)

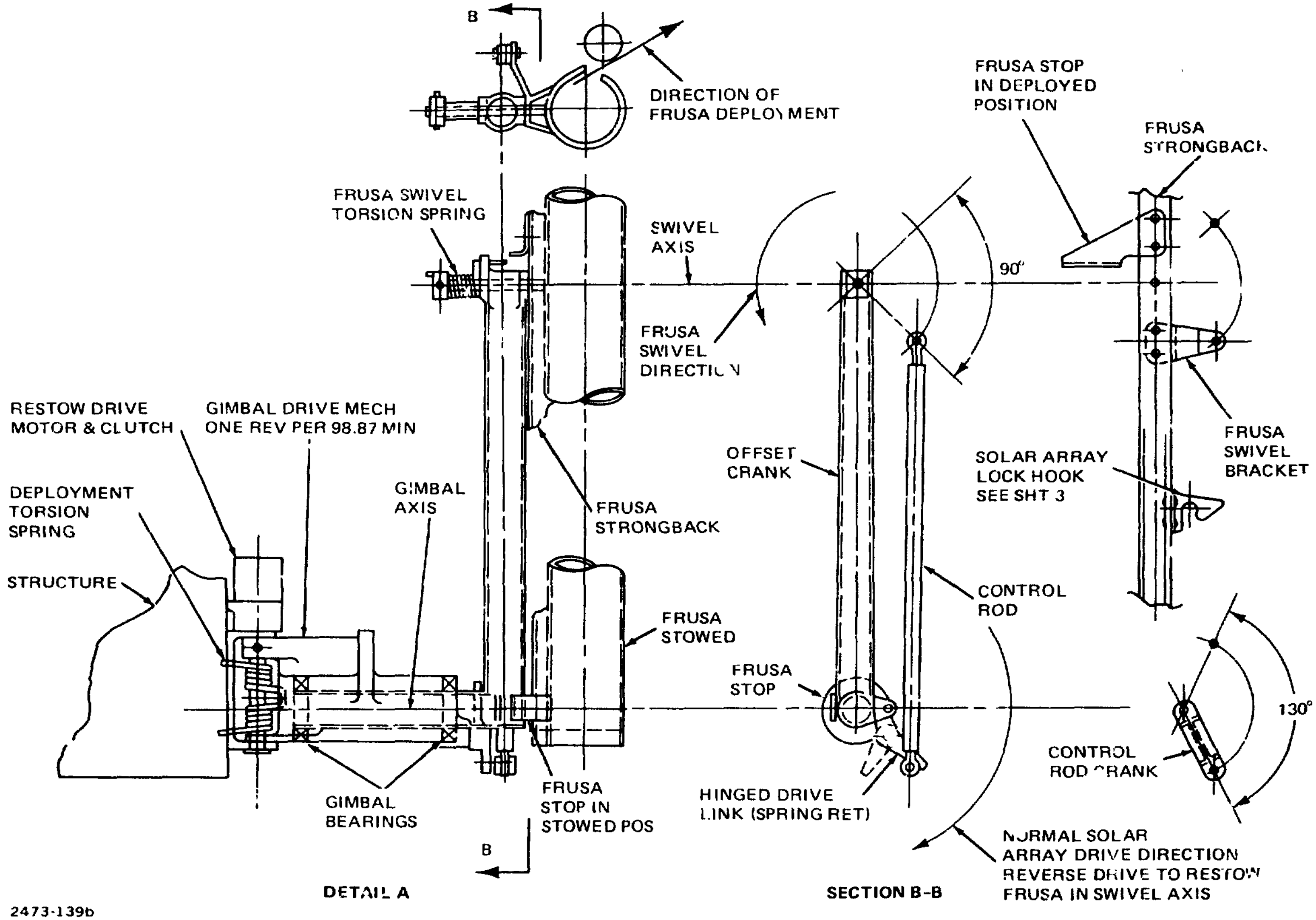


FIGURE 3.2-9 LANDSAT SOLAR ARRAY ASSEMBLY AND DETAILS (SHEET 2 OF 3)

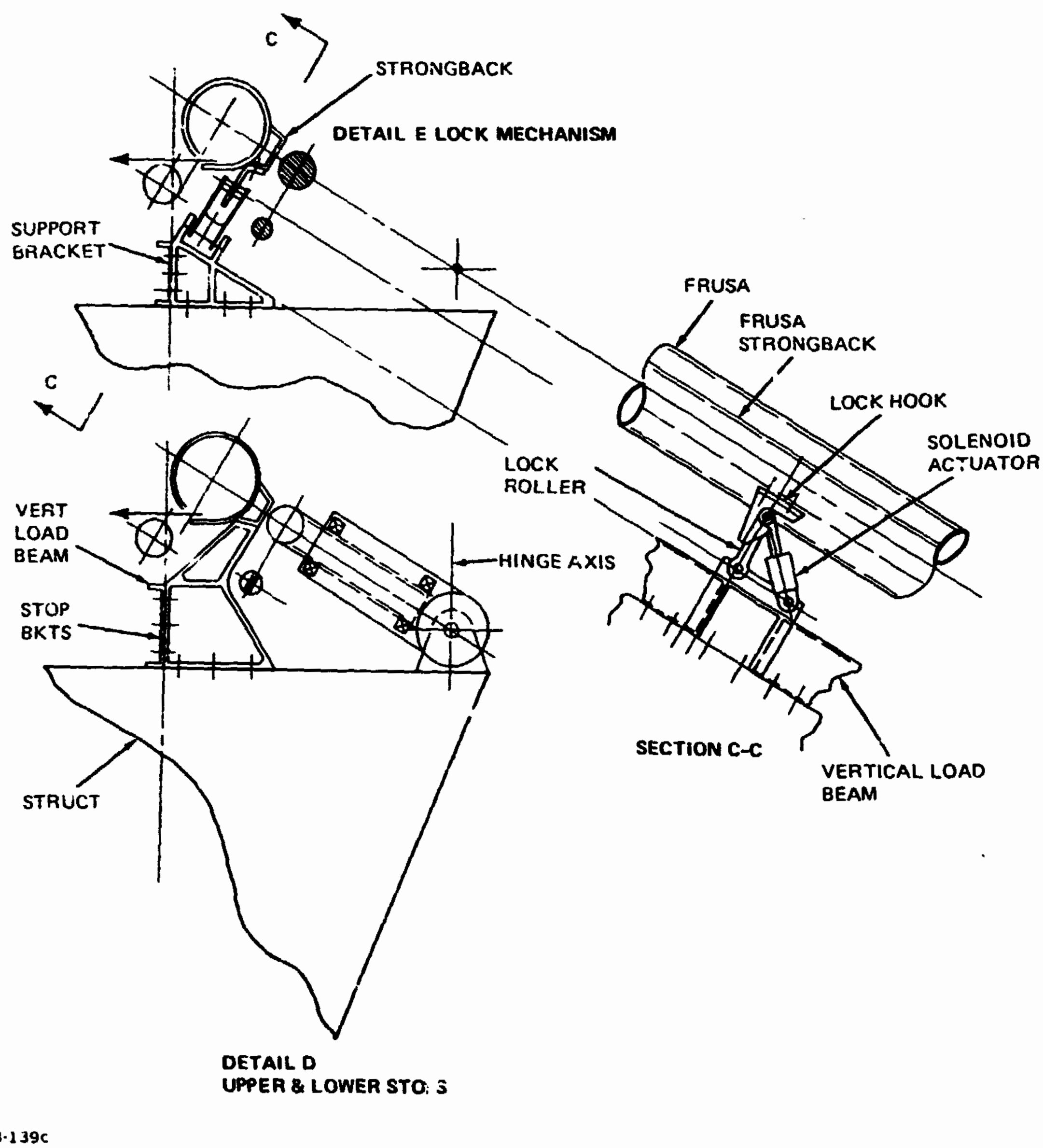


FIGURE 3.2-9 LANDSAT SOI AR ARRAY ASSEMBLY AND DETAILS (SHEET 3 OF 3)

After launch and insertion into orbit, the Solar Array will be deployed in the following sequence of operations:

(1) The lock hook is released by a solenoid actuator.

(2) After lock hook release, the deployment torsion spring drives the Solar Array components from their stowed position to the extended position, where the gimbal axis is in its correct orientation (time, 5 sec).

(3) Upon attainment of the correct gimbal axis orientation, the gimbal drive mechanism is activated at its maximum velocity of four times its normal velocity (normal velocity is one revolution in 98.87 min). This motion unlocks the FRUSA canister from its orientation parallel to the offset crank, and permits the FRUSA swivel torsion spring to rotate the canister 90° around the swivel axis to its correct orientation for the next step (time, approximately 8 min). The FRUSA drive continues at maximum velocity until it is in the proper orientation with the sun, at which time its velocity is reduced to normal velocity where it will remain for the life of the Landsat vehicle.

(4) With the FRUSA canister in its correct orientation about the swivel axis, deployment of the Solar Array is started. The array is unwound from its launch and storage drum to its operating position (time, approximately 5 min, total deployment time approximately 13 min).

This completes the extension cycle and would normally complete the mechanical requirements as well. However, for Landsat follow-on, an on-orbit resupply and return to earth requirement has been established. This requirement necessitates the restowage of the entire Solar Array system to its original launch configuration. It is because of this restow requirement that the deployment is not initiated with explosive components.

A brief description of the restowage sequence follows starting from a fully deployed operating position.

(1) The FRUSA Solar Array is rewound onto its drum inside the canister (time approximately 5 min).

(2) The Solar Array gimbal drive is reversed at its maximum velocity of four times normal. This motion causes the FRUSA canister to swivel 90° to its stowed position, parallel to the crank arm and comes to rest with the crank arm in the correct orientation for the next step in the restowage sequence. The action of swiveling the FRUSA canister also preloads the swivel torsion spring, storing energy for the next deployment sequence (time approximately 8 min).

(3) The final step in the restowage sequence is the rotation of the entire drive mechanism, offset crank, and the FRUSA canister about the hinge axis as a unit until contact is made with the upper and lower stop brackets. Then, the locking hook mechanism snaps into the final locked position. This motion is accomplished by actuating the restow drive motor and clutch, causing the required motion about the hinge axis and simultaneously preloading the deployment torsion spring for the next deployment sequence (time approximately 5 min, total restowage time approximately 18 min).

3.2.2 TDRS Antenna

Much of the positional geometry for the TDRS antenna was determined in the study of the Solar Array configuration since the two subsystems are very much interrelated with respect to line-of-sight geometry and obscuration problems.

3.2.2.1 General Configuration

Since antenna location and deployed position have been defined by the earlier study, an investigation was initiated to define and/or optimize the stowed position and the deployment components. The stowed position for the 76-in. diameter antenna dish in the T-1A configuration requires the least complex mechanism because it appears that a fixed geometry knee joint could be used. If the required stiffness could be obtained in both the stowed and deployed condition, then, the mechanical simplicity of this scheme would make it a very desirable design. Since the deployed length of the antenna mast required for the 0930 launch is so short, the need for the Astromast design with its large diameter and extension capability is avoided (The Astromast design was an early requirement established by NASA/GSFC to accommodate a range of launch times up to 1100 hours. See Figure 3.2-10). Further study of the deployment geometry for the T-1A configuration determined

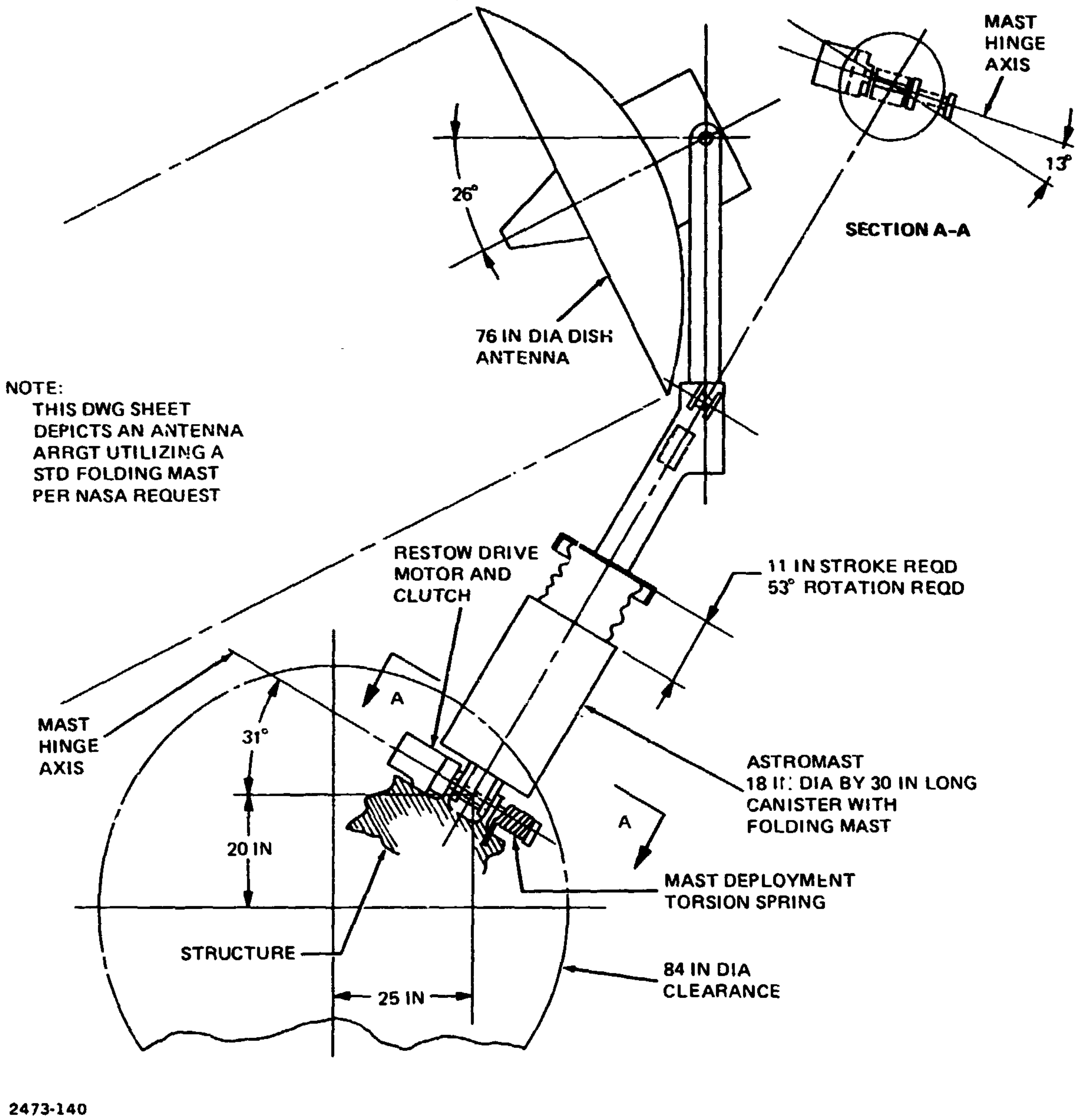


FIGURE 3.2-10 T-1A CONFIGURATION–TDRS ANTENNA ASSEMBLY AND DETAILS

that the stowed length and the deployed length of the lower section of the antenna mast differ by only 11 inches. Furthermore, no angular changes are necessary between the upper and lower sections of the antenna mast during deployment. Since the length change was so small, it was possible to design a simple telescoping mast and eliminate the need for the Astromast design. Design details for the simple telescoping mast is shown in Figure 3.2-11. The H-1A configuration can be satisfied by a similar design if the stowed position of the dish antenna can be made to duplicate the T-1A configuration. Figures 3.2-12A, -12B, and 3.2-13 define the H-1A and T-1A geometry and mechanical details sufficiently well to prove the suitability of the canted antenna stowed position. While the 50-inch mast extension required is significantly greater than that required for the T-1A configuration, Figure 3.2-13 clearly shows that it can be accomplished with a single telescoping tube arrangement. Because of the longer deployed length of the antenna mast, larger diameter tubes have been selected to insure that the natural frequency in the deployed mode equals or exceeds 1 to 2 cycles per second.

3.2.2.2 Gimbal Lock Mechanism

Since the antenna gimbal drives would be required to withstand launch loads and at the same time hold the dish rigidly in the stowed position, it becomes obvious that some type of gimbal lock mechanism would be required. Although this detail of mechanism design is beyond the scope of the present contract, a sketch of a possible mechanical arrangement that would be operated by the telescoping action of the antenna mast during deployment was prepared. A schematic arrangement is shown in Figure 3.2-14. This arrangement is compatible with the antenna mast system for either the T-1A or H-1A configuration.

3.2.2.3 Antenna Deployment and Restowage

The deployment sequence of the antenna system is extremely simple. After launch and insertion into orbit, the lock hook is released by a solenoid actuator. Subsequent to lock hook release, the deployment torsion spring drives the antenna system components about the hinge axis, from their stowed position, to the extended position in approximately 10 seconds. During deployment, the shrink cable geometry permits the extension spring to extend the mast. During mast extension, the outboard section of the mast is caused to swivel 53° by a cam built into the mast to provide proper alignment of the antenna mast components in the deployed position.

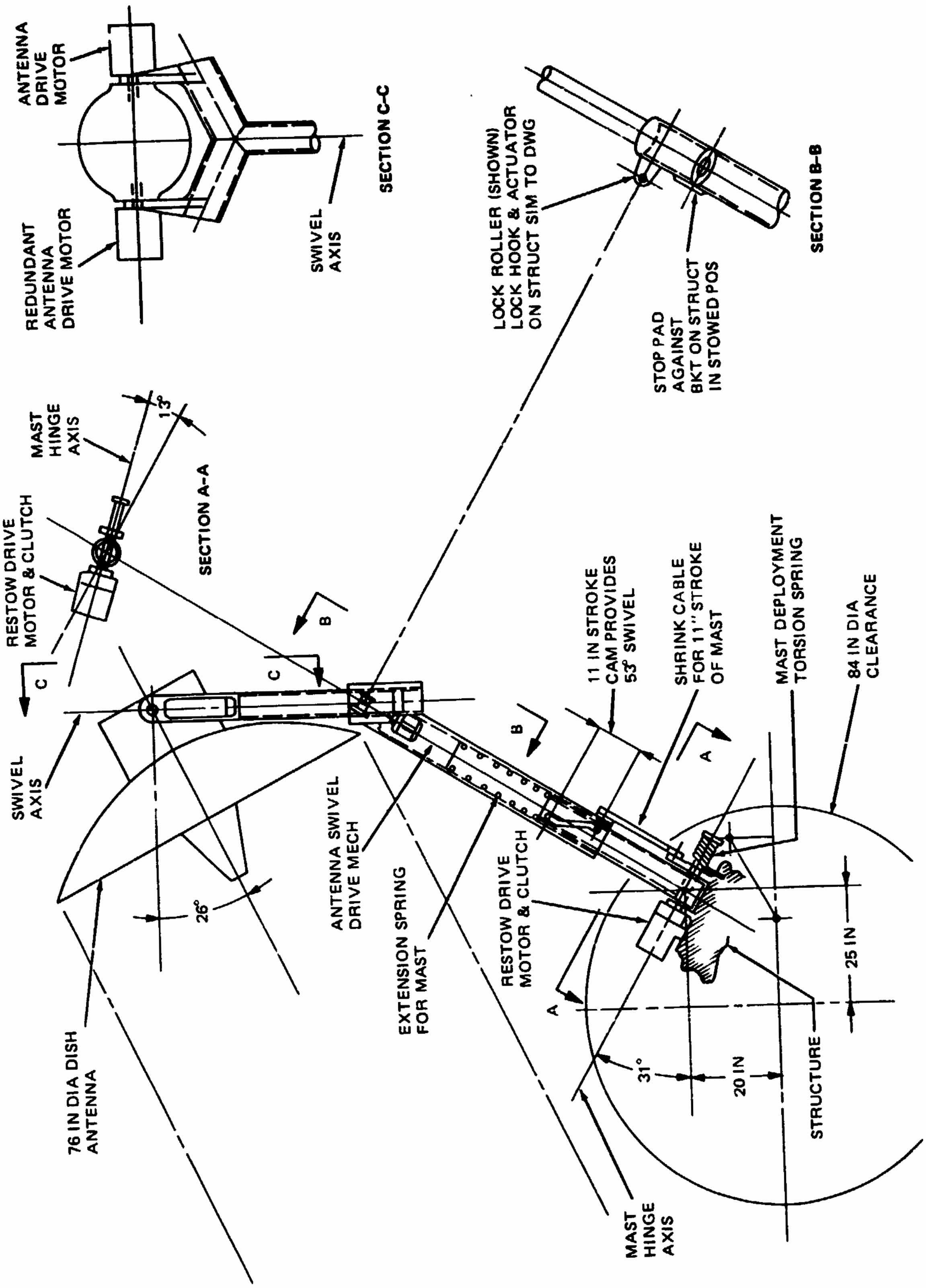


FIGURE 3.2-11 T-1A CONFIGURATION—TDRS ANTENNA ASSEMBLY AND DETAILS

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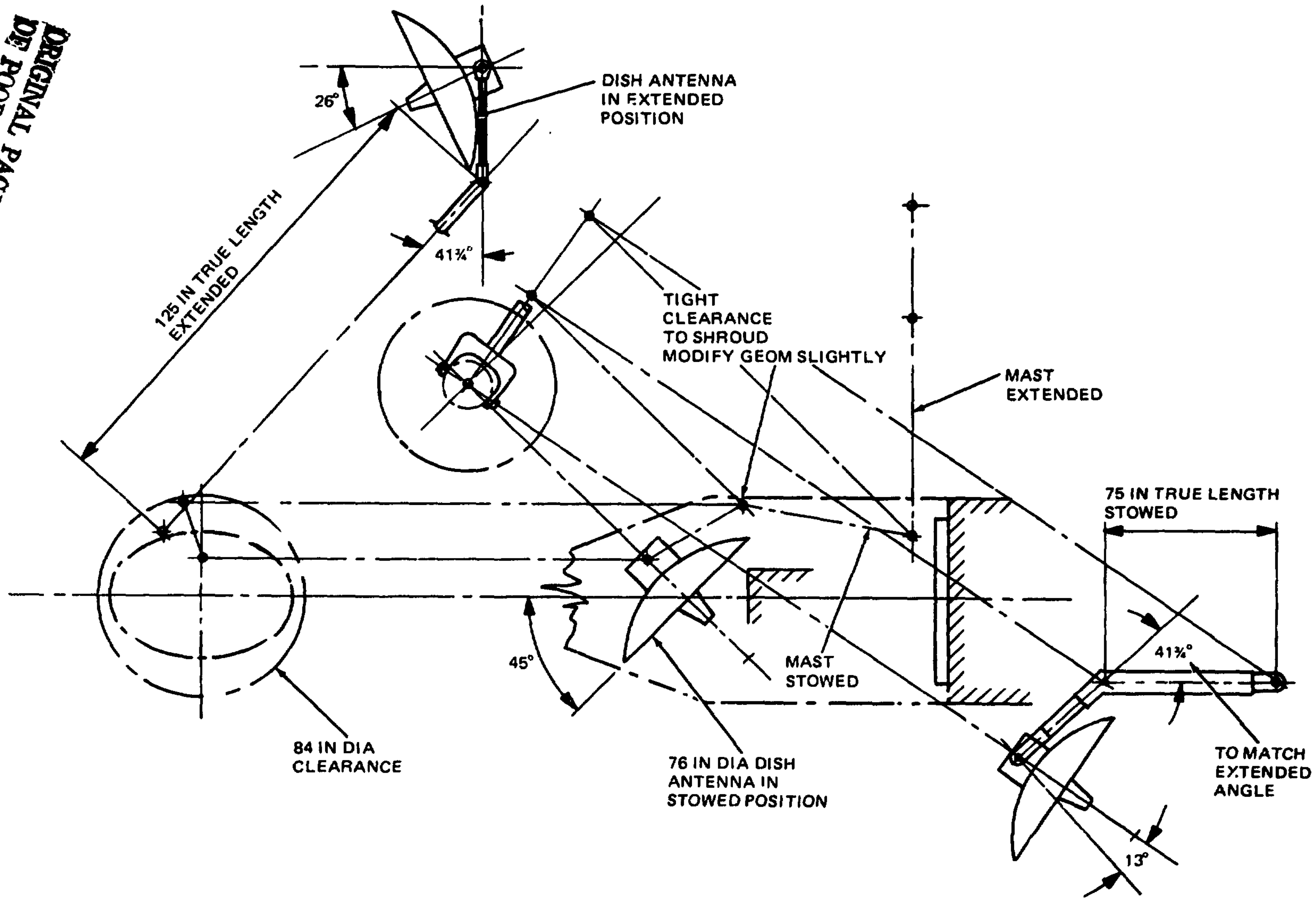
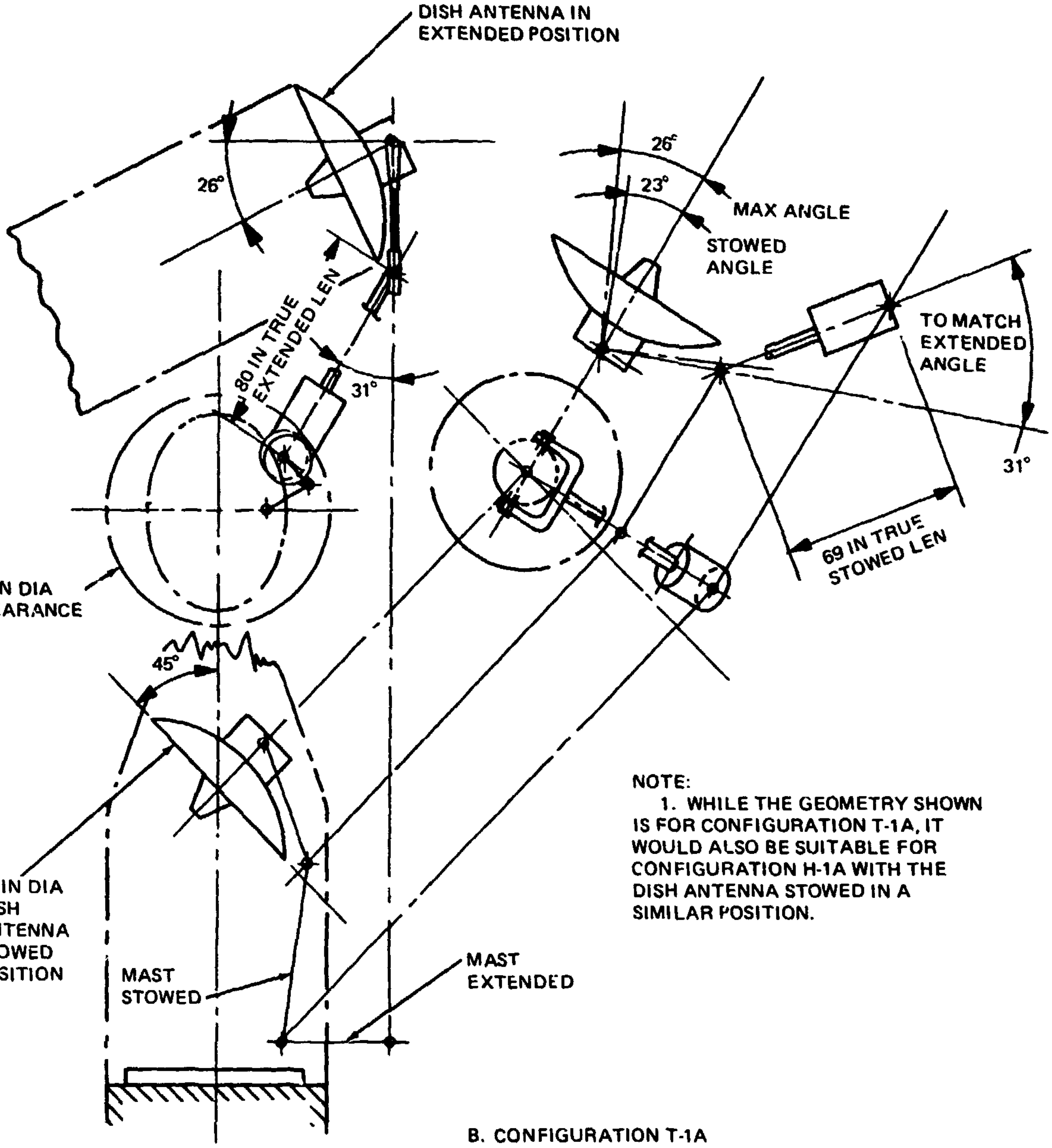


A. CONFIGURATION H-1A

2473-143a

FIGURE 3.2-12A TDRSS ANTENNA DEPLOYMENT GEOMETRY



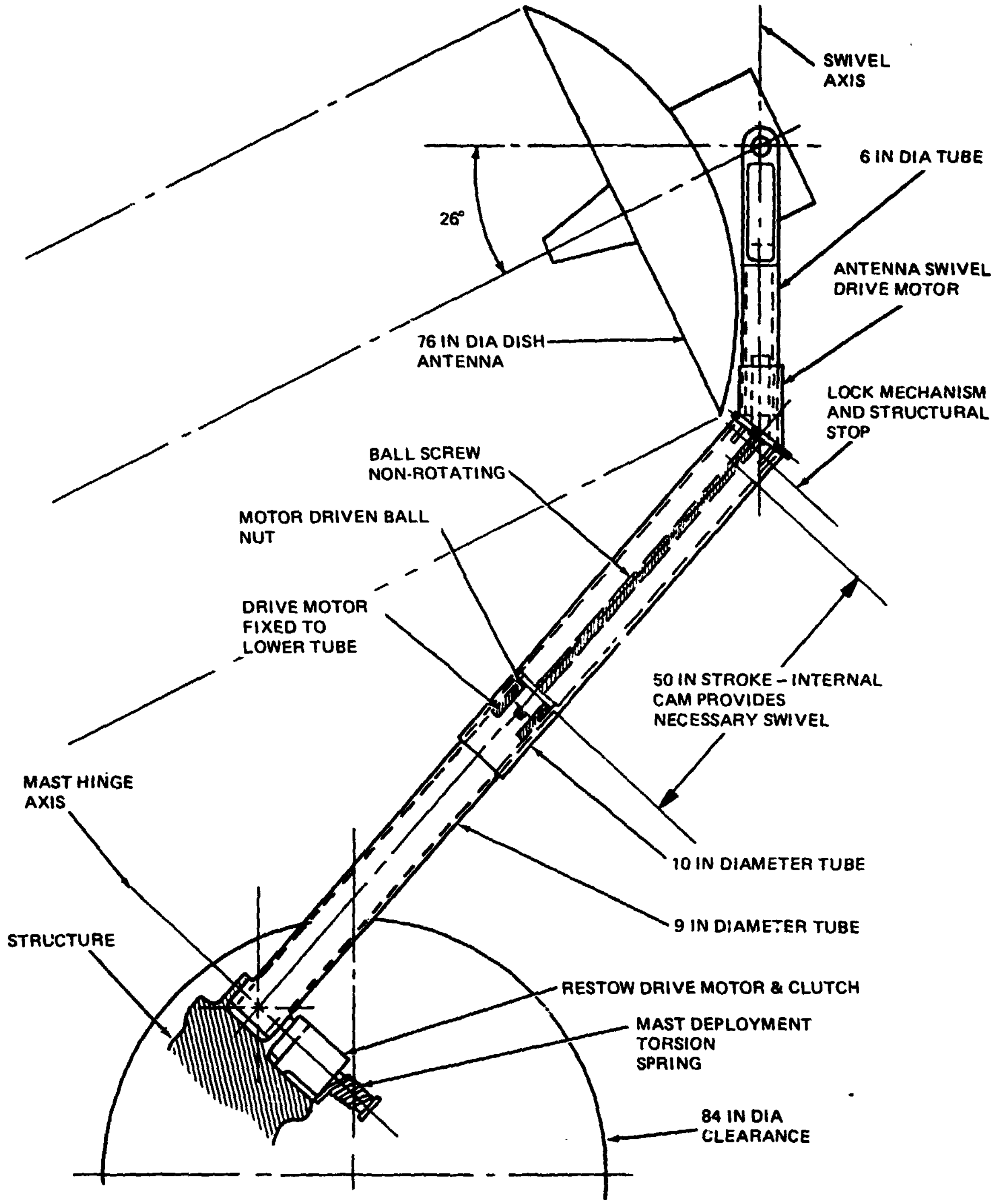
NOTE:

1. WHILE THE GEOMETRY SHOWN IS FOR CONFIGURATION T-1A, IT WOULD ALSO BE SUITABLE FOR CONFIGURATION H-1A WITH THE DISH ANTENNA STOWED IN A SIMILAR POSITION.

B. CONFIGURATION T-1A

2473-143b

FIGURE 3.2-12B TDRSS ANTENNA DEPLOYMENT GEOMETRY



2473-144 FIGURE 3.2-13 H-1A CONFIGURATION ANTENNA ASSEMBLY AND DETAILS

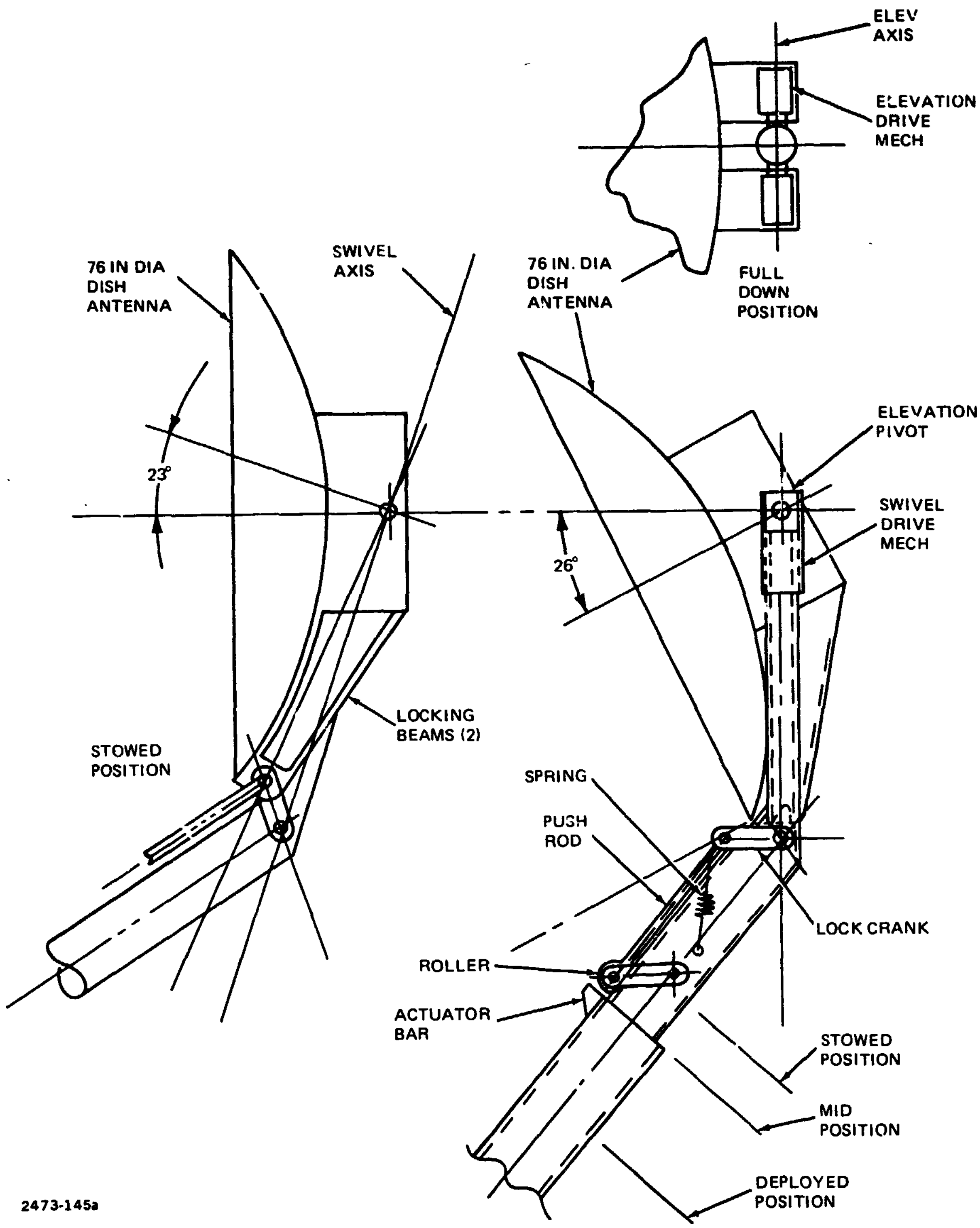
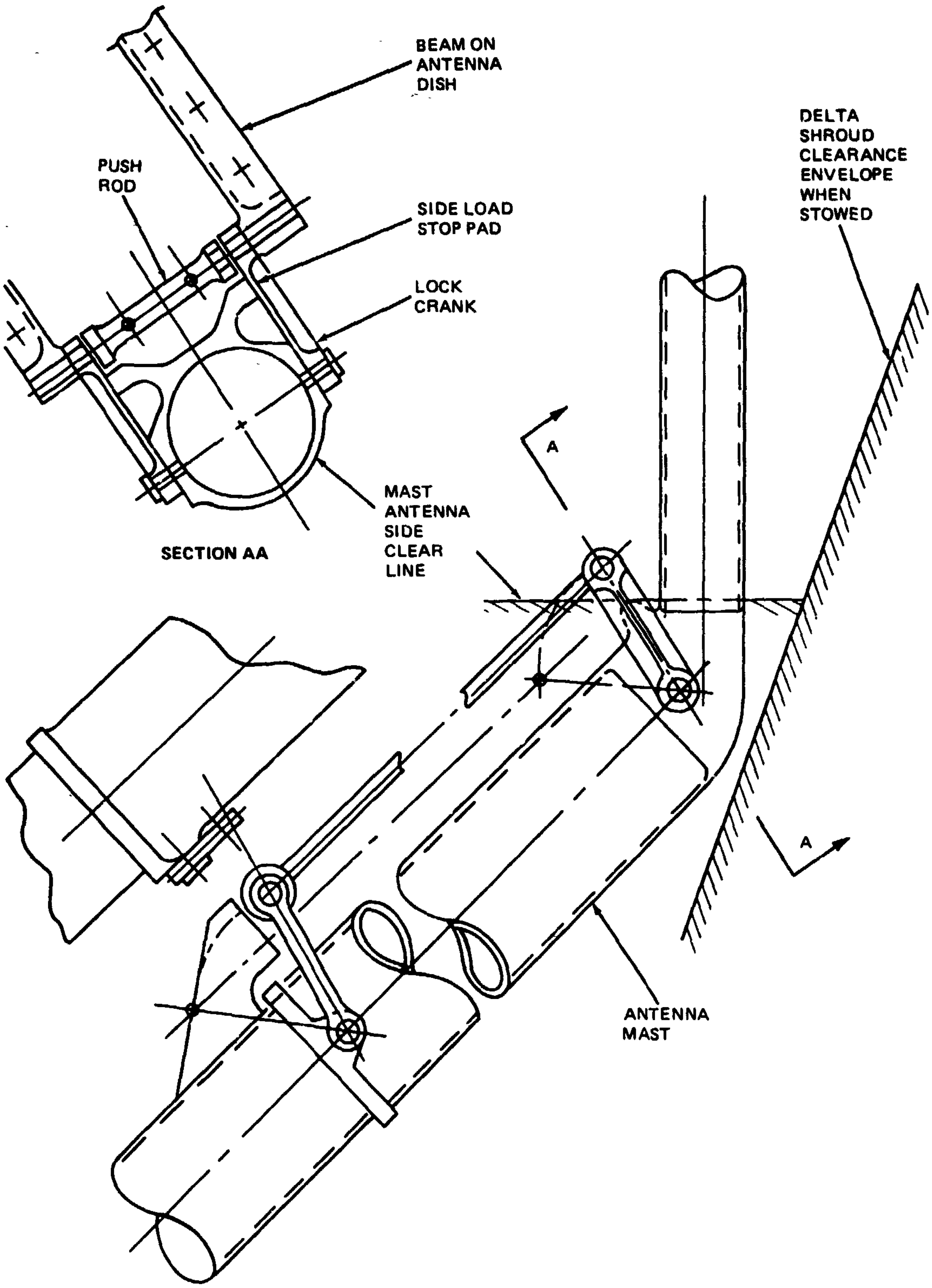


FIGURE 3.2-14 GIMBAL LOCK MECHANISM (SHEET 1 OF 2)



2473-145b

FIGURE 3.2-14 GIMBAL LOCK MECHANISM (SHEET 2 OF 2)

The dish antenna can now be oriented as desired to meet mission requirements. The restow sequence is slightly more complex and a brief description follows:

(1) The dish antenna is rotated to the correct angle for retraction in azimuth.

(2) The dish antenna is rotated to 23° below the horizontal in elevation.

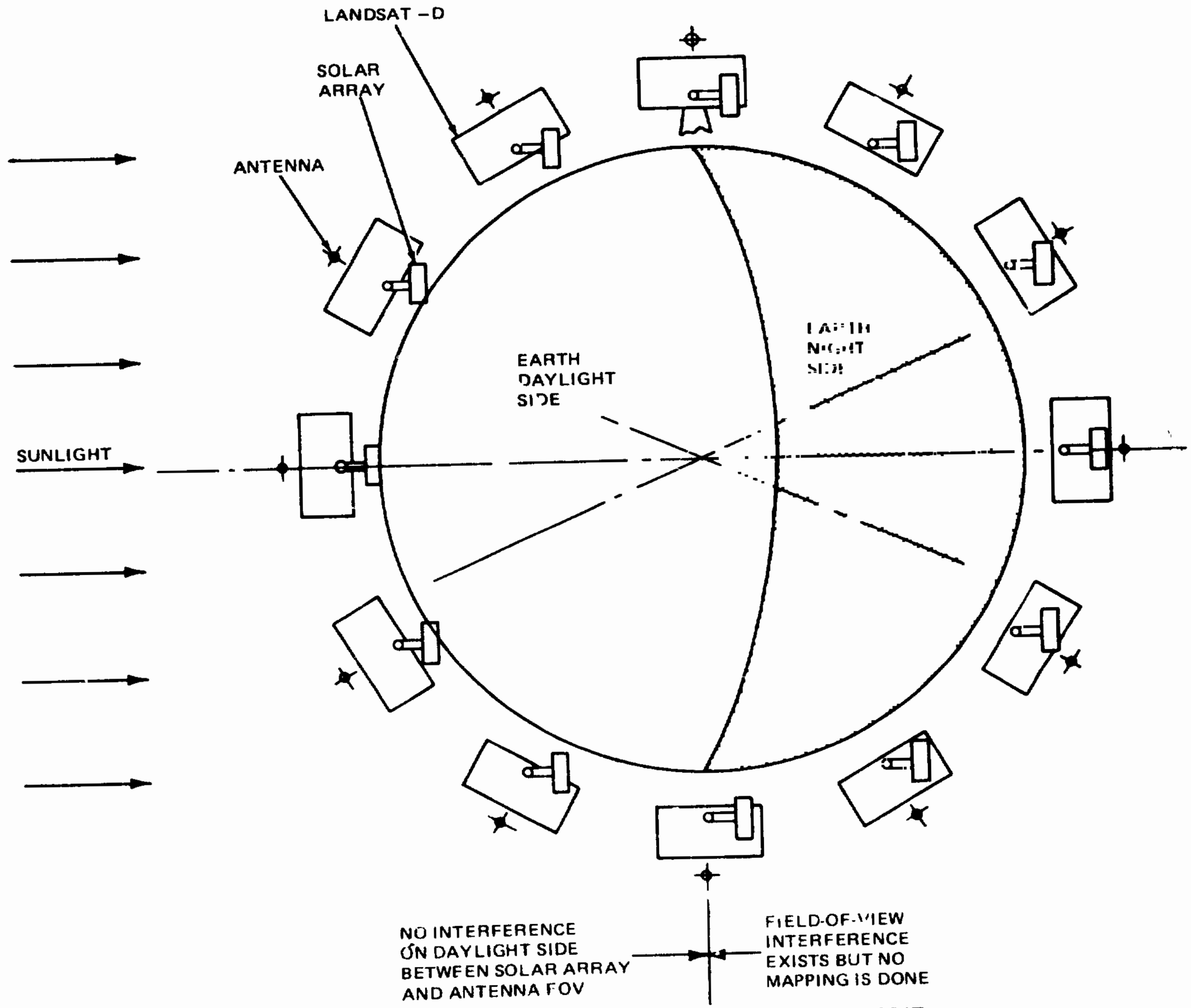
(3) The restow drive motor and clutch can now be actuated, causing the required motion about the hinge axis until contact is made with the stop bracket and the locking hook mechanism snaps into the final locked position. During hinge axis rotation, the deployment torsion spring is preloaded for the next deployment. The shrink cable causes the mast to shorten by the required 11 inches, and preloads the extension spring while the cams cause the required 53° mast rotation. The time required for restowage about the hinge axis will be approximately five minutes.

3.2.2.4 Alternate Launch Time

Early in Phase I, the launch time was established to be 1100 hours. Subsequently, the launch time was changed to 0930 hours. As a consequence, all of the Solar Array and TDRS antenna studies for Concept Validation have been based on the 0930 launch time. The Solar Array orientation during orbit is shown in Figure 3.2-15.

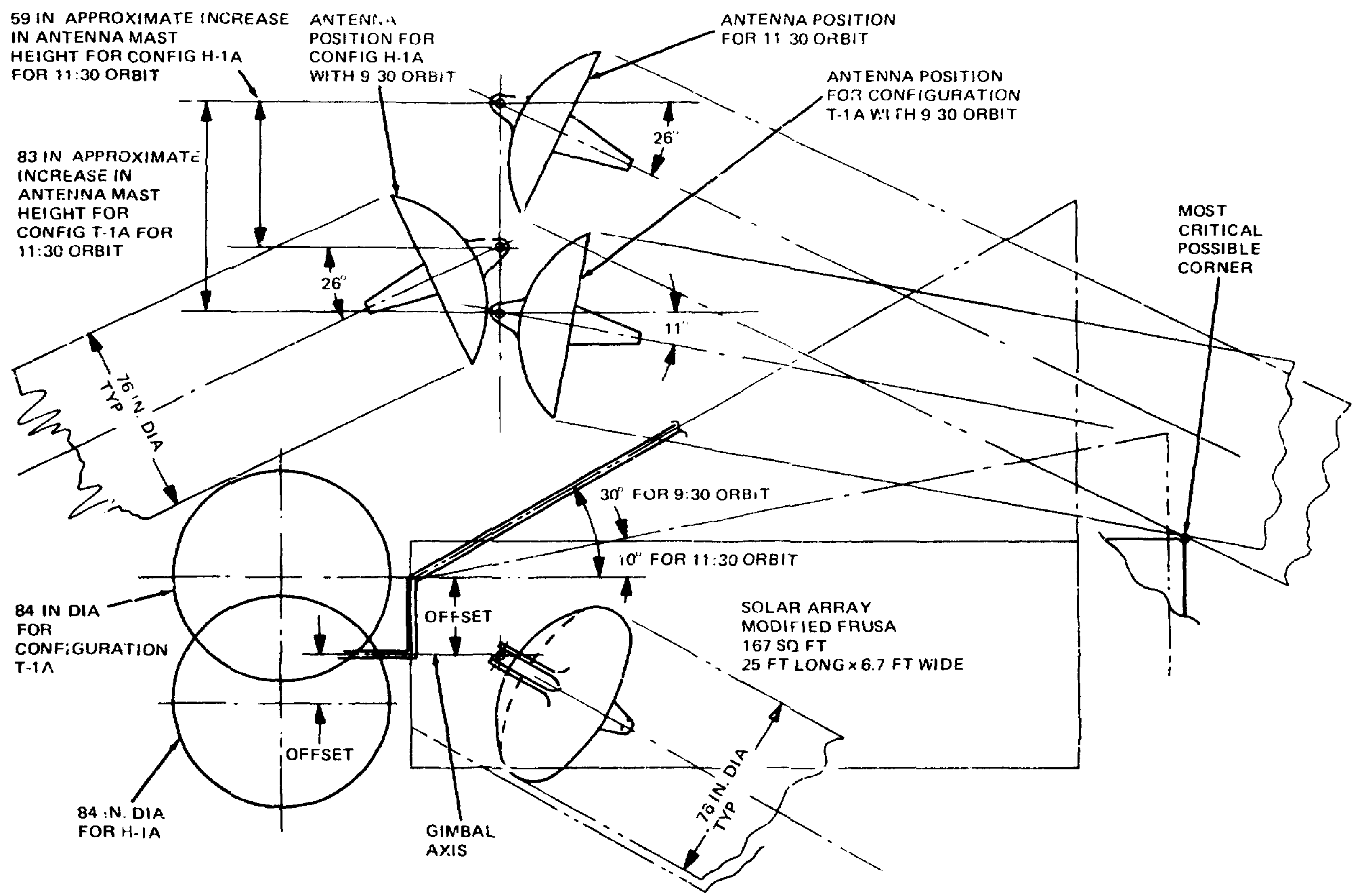
Since this parameter governs the orbit plane relative to incident solar radiation, the orientation of the Solar Array must also be changed for efficient collection of solar energy. The change in solar array orientation associated with later launch times, however, intrudes into the TDRS antennas clear line-of-sight during portions of the satellite operation. To recover the line-of-sight geometry, it is necessary to increase the distance separating the TDRS antenna and the Solar Array. This is best accomplished by an increase in antenna mast length.

For the extreme case of a 1130 launch, the T-1A configuration antenna mast length must be increased by 83 inches over that required for the 0930 launch. Similarly, the H-1A antenna mast length must be increased by 59 inches for the 1130 launch (See Figure 3.2-16).



2473-145

FIGURE 3 2-15 SOLAR ARRAY ORIENTATION DURING ORBIT



2473-147

FIGURE 3.2-16 TDRS ANTENNA LOCATION FOR 11:30 ORBIT

3.2.3 Module Exchange Concept

One of the Landsat follow-on mission requirements specifies an on-orbit equipment resupply capability. This is to be accomplished by using the Shuttle and a special purpose Module Exchange Mechanism (MEM) for remote manipulation of equipment packages.

3.2.3.1 MEM Adapter

The MEM being developed by Rockwell International is designed primarily for components larger than that found in the Landsat vehicle. The manipulator is designed to operate with two pickup points, but only with pickup spans of 40 inches or greater.

Since none of the Landsat components can satisfy this large span requirement, it is recommended that an adapter to the MEM be used. This adapter will utilize only one mechanized pickup point operated by an electrical drive system. A feature of this adapter is that no direct high axial loads are required to remove or install equipment. A schematic arrangement for the MEM adapter is shown in Figure 3.2-17. Figures 3.2-18A through -18C show some alternate concepts for MEM.

3.2.3.2 Equipment Mounting

As a supplemental design investigation to the provisions for on-orbit module exchange, a concept for equipment mounting was devised for the Landsat equipment.

The mounting scheme is based on a design that employs one bolted joint and two other joints that are constrained but not rigidly fastened. This arrangement achieves a determinate load distribution that is capable of supporting all loading conditions that are anticipated; while at the same time, it is virtually insensitive to the dimensional changes normally associated with thermal gradients.

Equipment installation is compatible with a procedure that requires motion parallel to the mounting plane, which is the requirement of the Instrument Module design. A schematic arrangement for the mount assembly and details is shown in Figure 3.2-19. Figure 3.2-20 shows an alternate equipment mounting scheme.

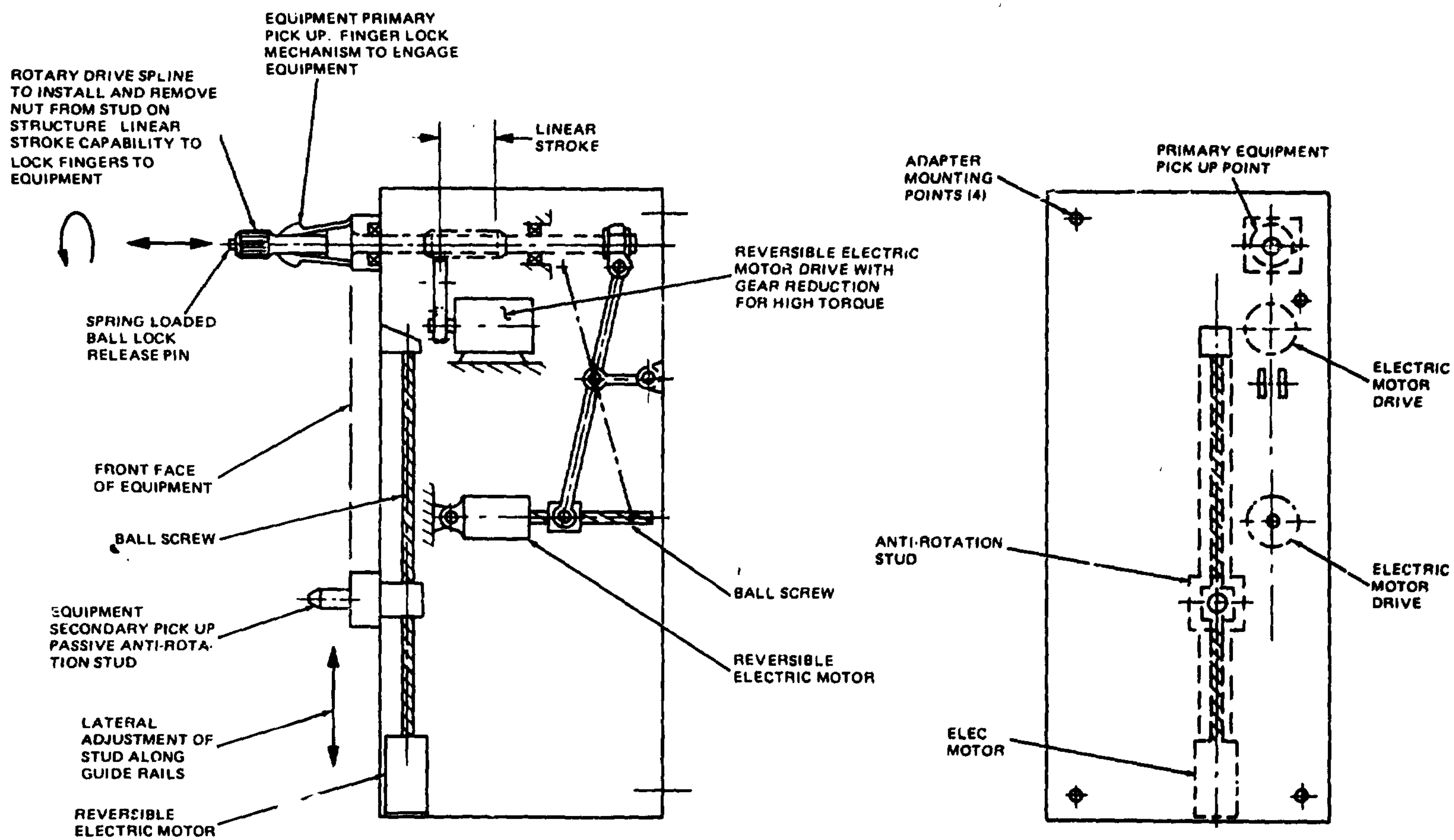


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FIGURE 3.2-17 SCHEMATIC ARRANGEMENT FOR MODULE EXCHANGE MECHANISM (MEM) ADAPTER

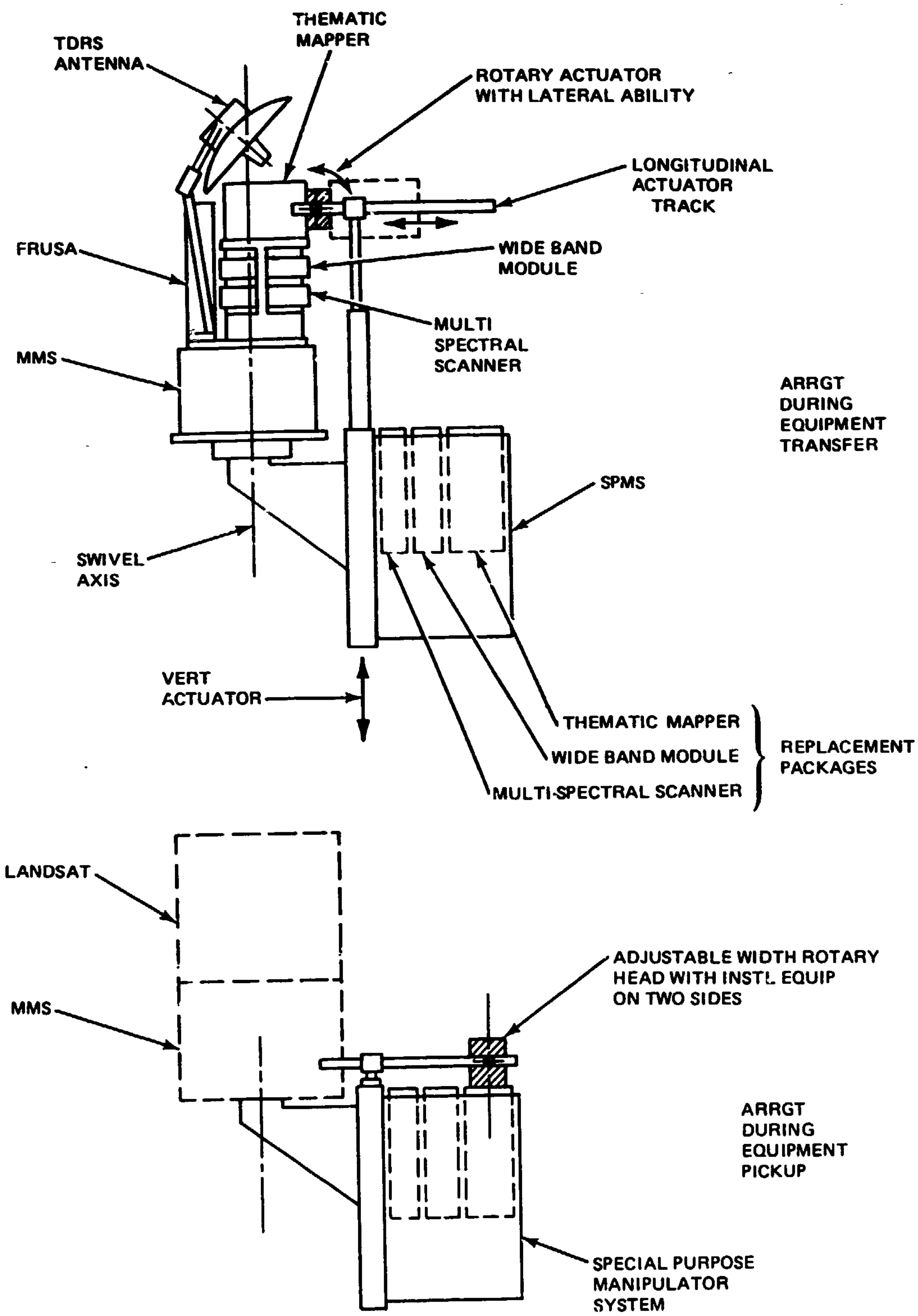

FIGURE 3.2-18A ALTERNATE CONCEPTS FOR MODULE EXCHANGE MECHANISM ADAPTER (MEM)

2473-159

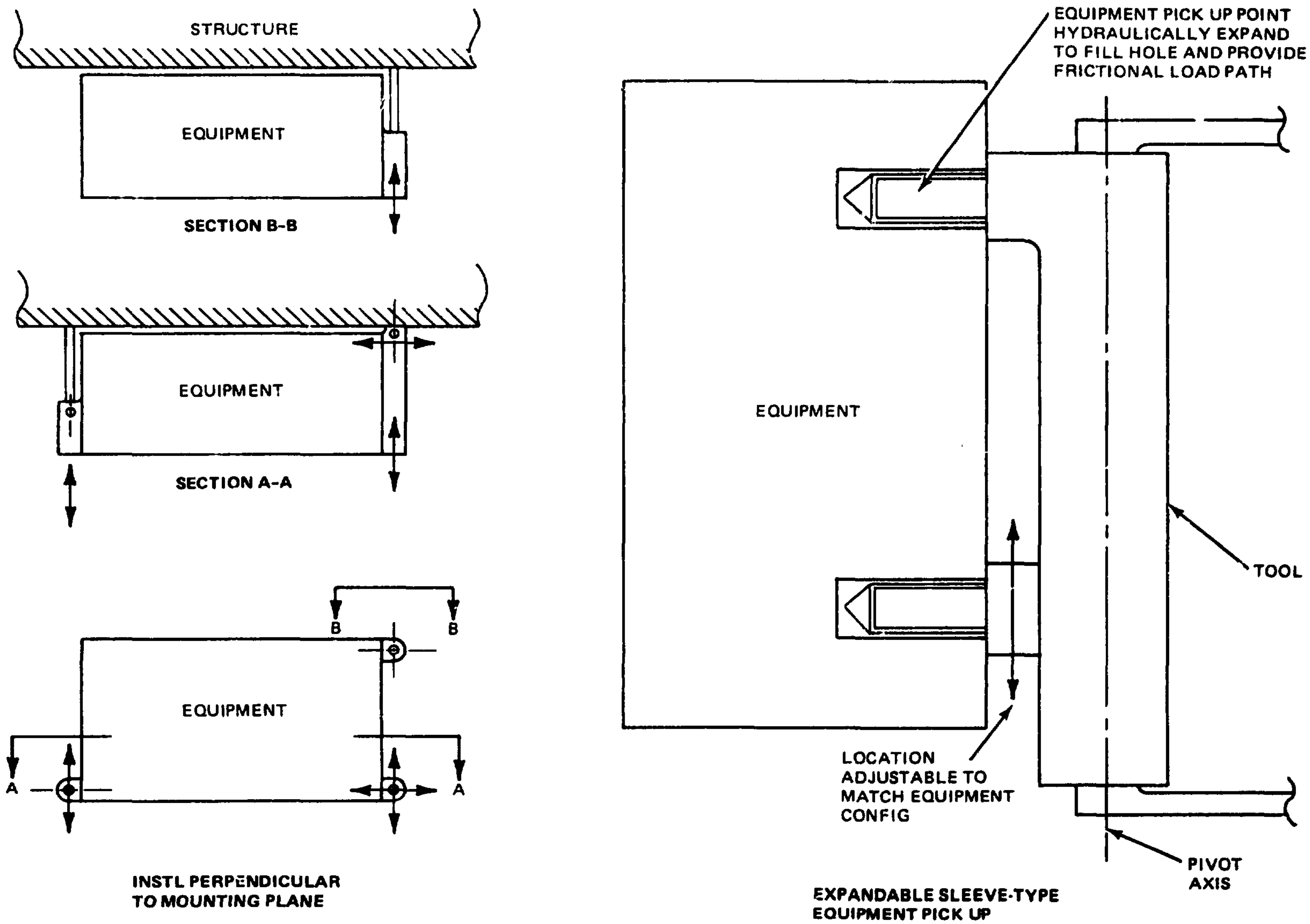

FIGURE 3.2-18B ALTERNATE CONCEPTS FOR MODULE EXCHANGE MECHANISM ADAPTER (MEM)

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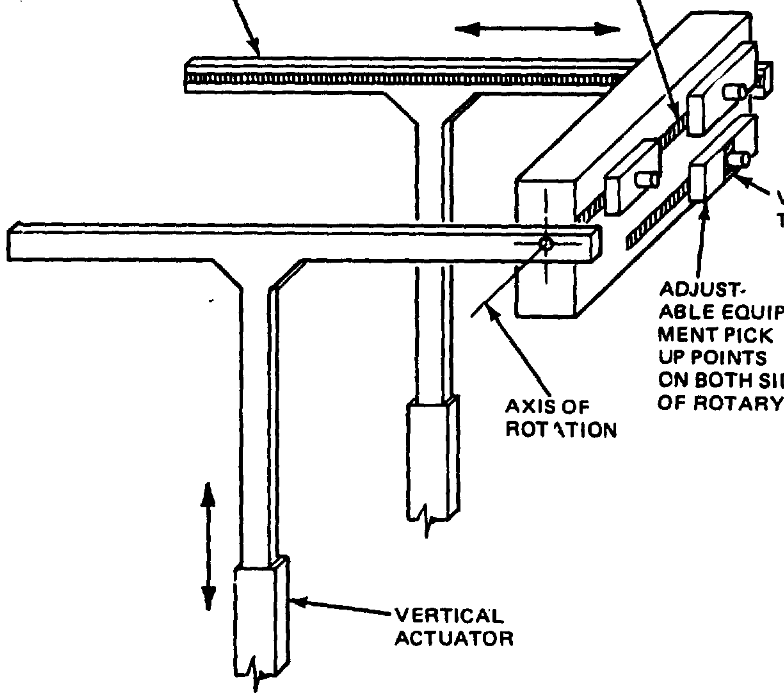
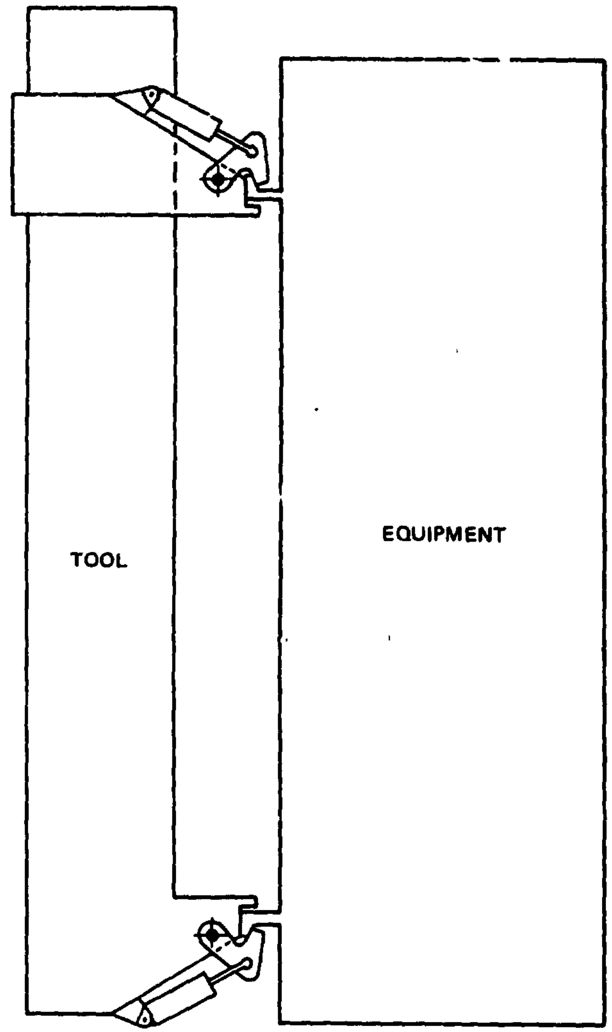


SPECIAL PURPOSE MANIPULATOR SYSTEM

UPPER END ARRGT



SUITCASE LATCH-TYPE EQUIPMENT PICK UP CONFIGURATION

FIGURE 3.2-18C ALTERNATE CONCEPTS FOR MODULE EXCHANGE MECHANISM ADAPTER (MEM)

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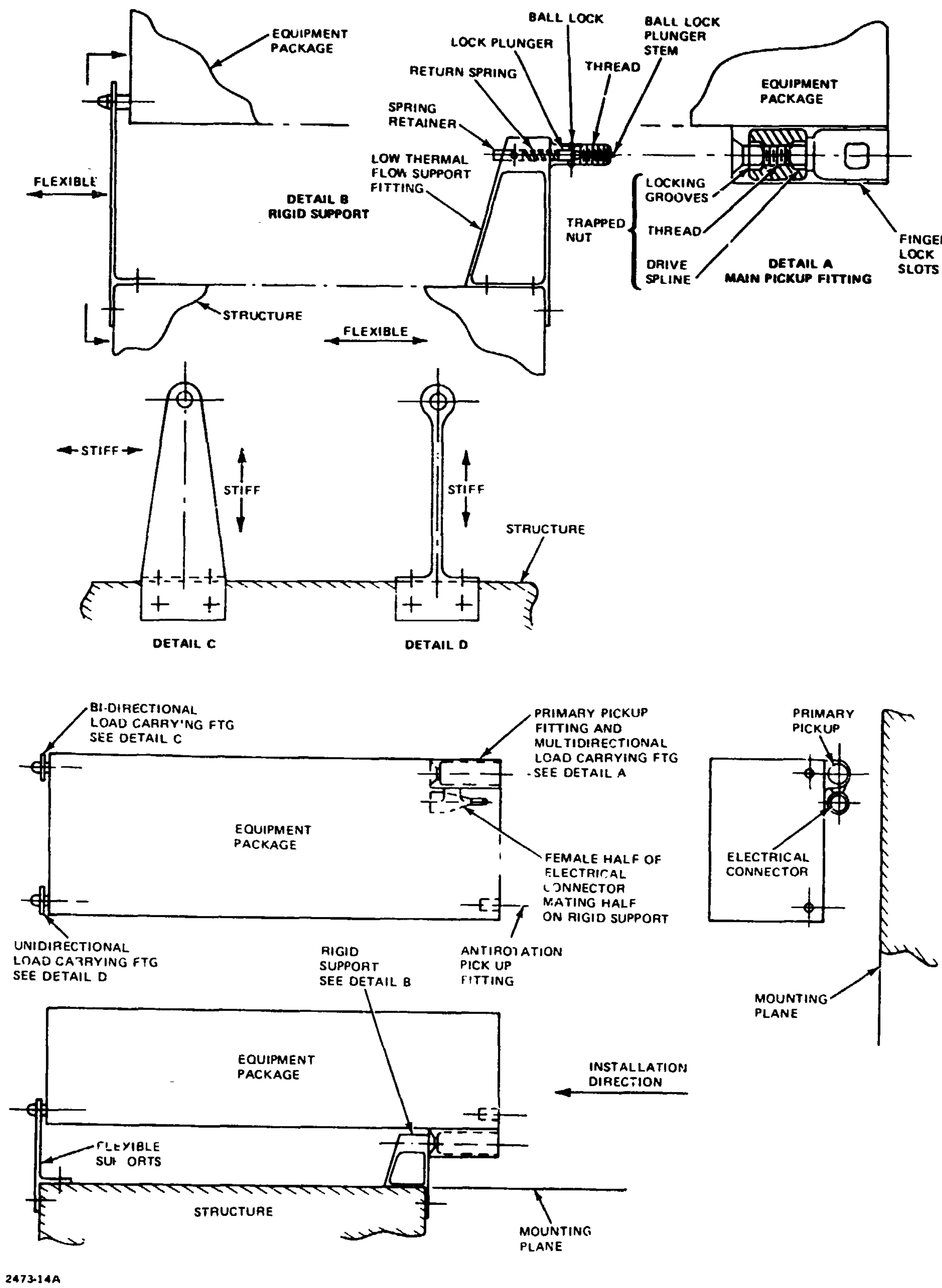

FIGURE 3.2 19 EQUIPMENT MOUNTING ASSEMBLY AND DETAILS (SHEET 1 OF 3)

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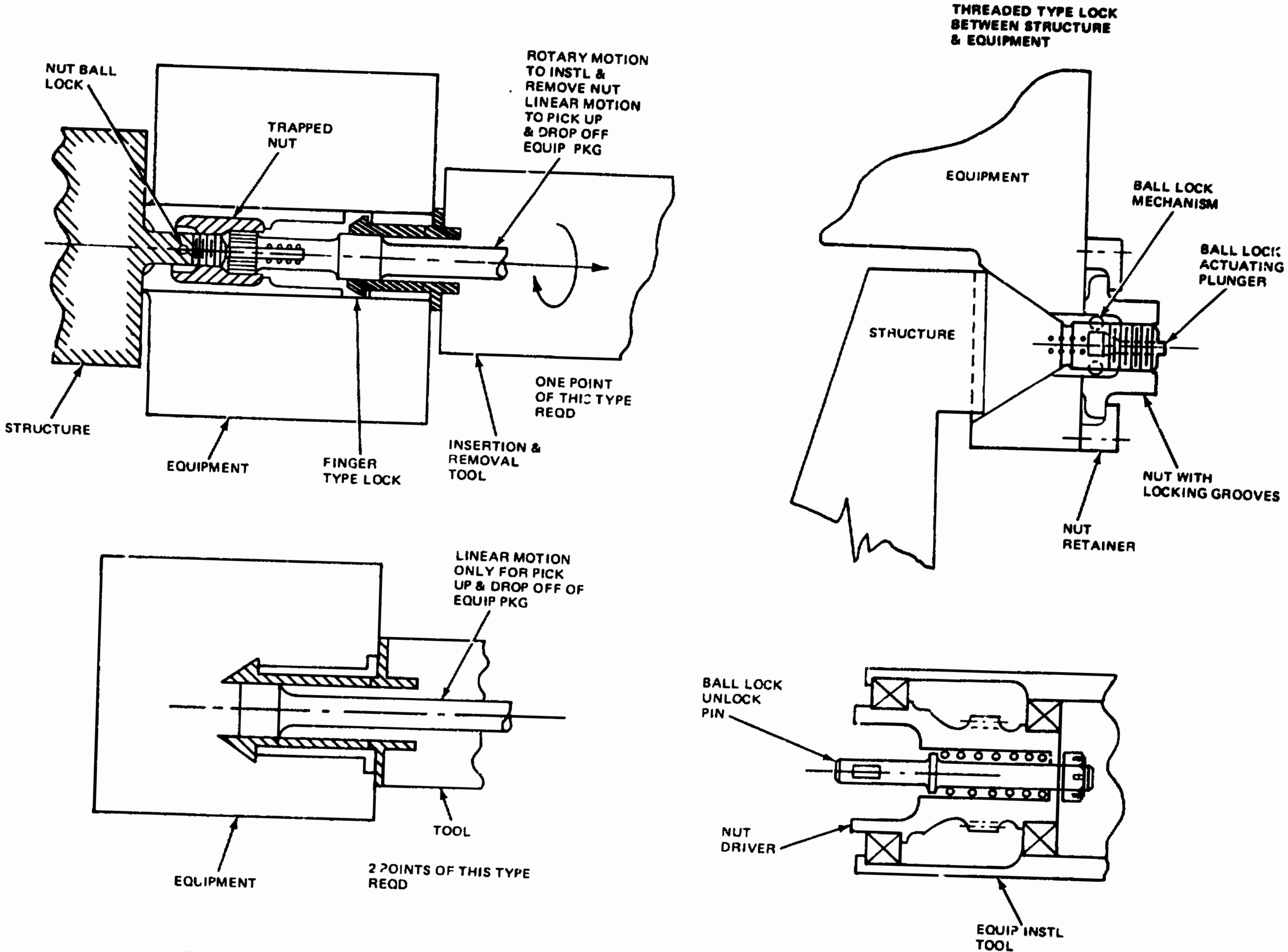


FIGURE 3.2-19 EQUIPMENT MOUNTING ASSEMBLY AND DETAILS (SHEET 2 OF 3)

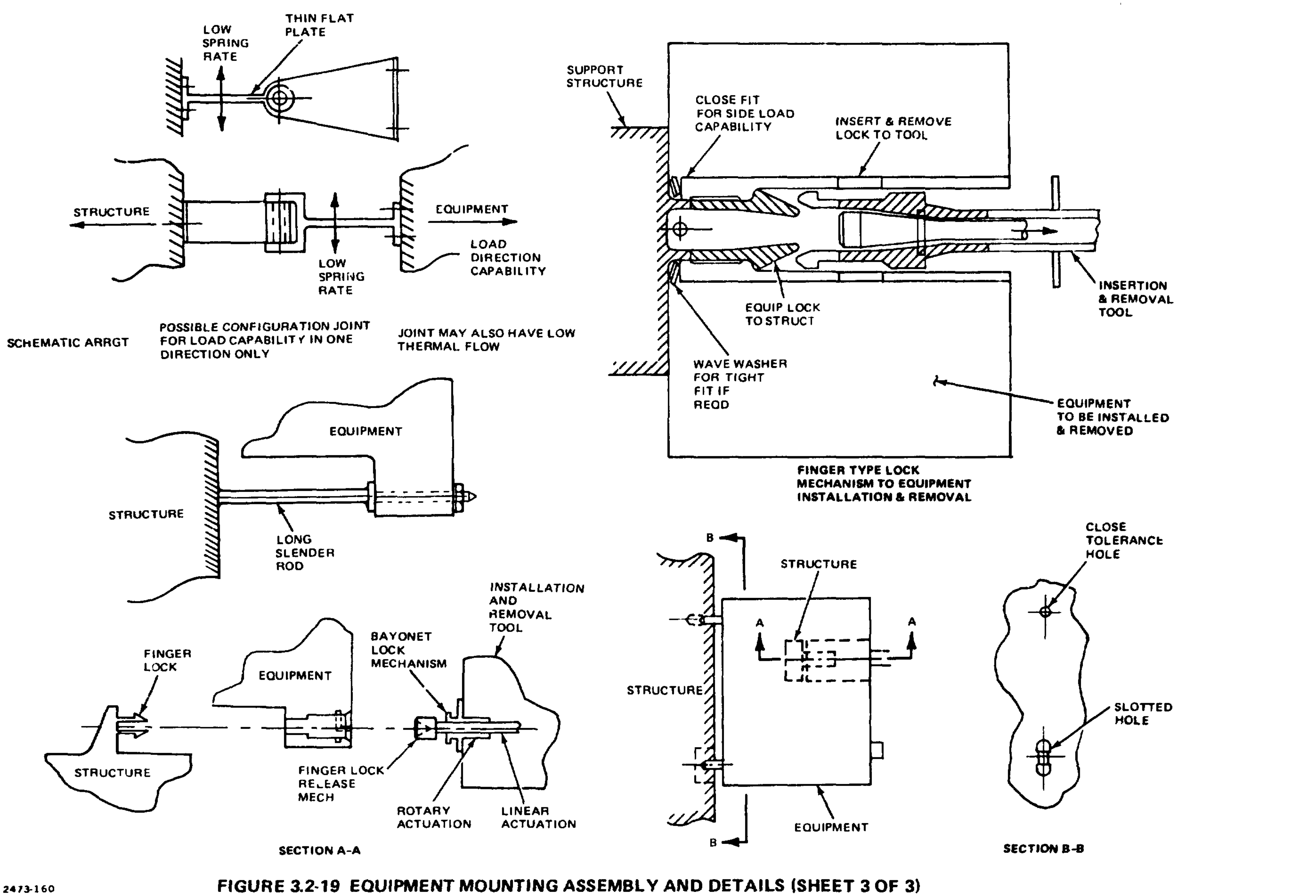


FIGURE 3.2-19 EQUIPMENT MOUNTING ASSEMBLY AND DETAILS (SHEET 3 OF 3)

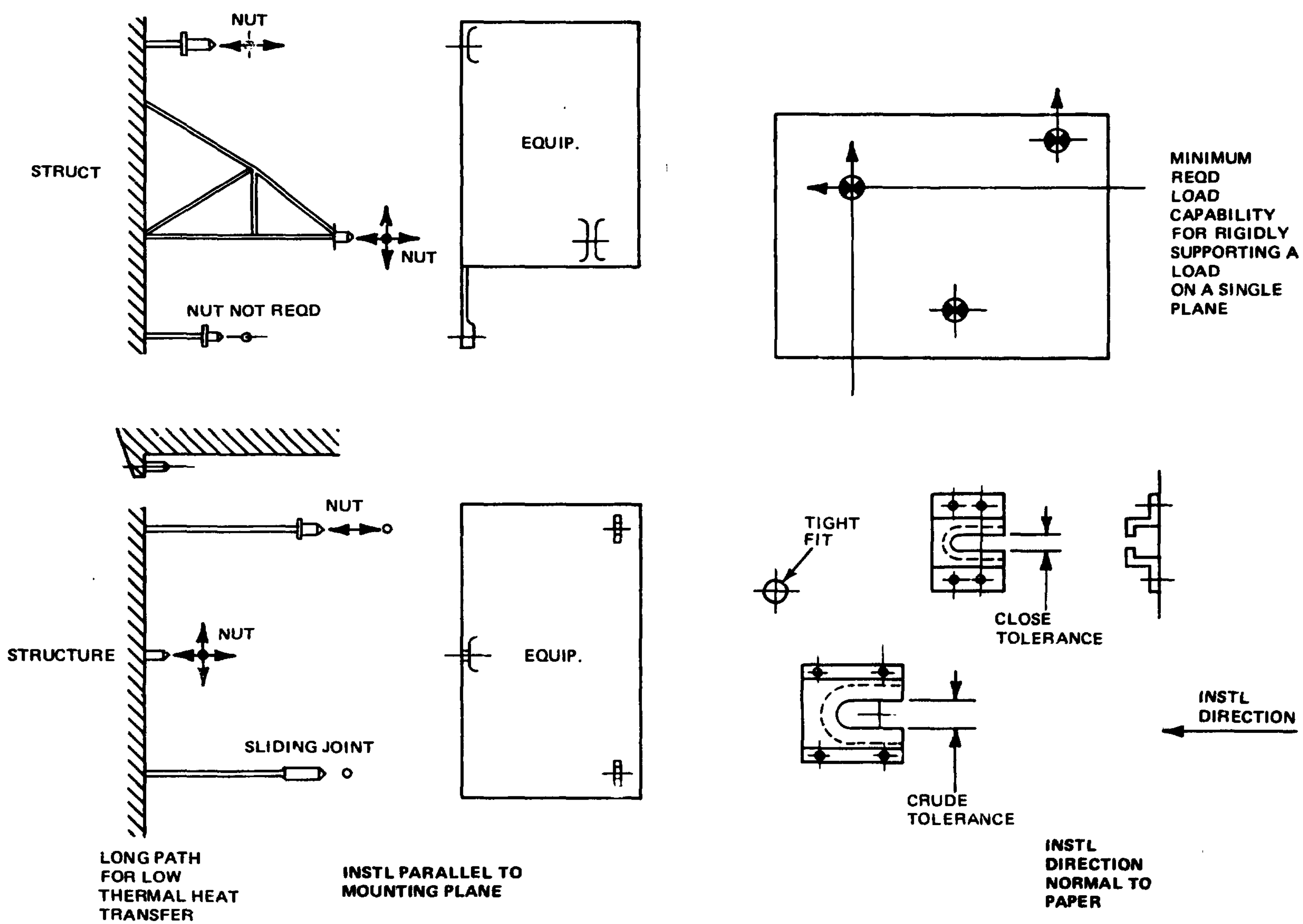


FIGURE 3.2-20 ALTERNATE CONCEPT FOR EQUIPMENT MOUNTING

2473-159

3.2.3.3 Emergency Disconnect

Early in the Landsat study, a requirement was established for an emergency disconnect system for the Solar Array and TDRS antenna subsystems. The intent of this requirement was to provide a jettison capability for these two large appendages should an emergency arise during or after deployment.

Figure 3.2-21 schematically shows one possible arrangement of a pyrotechnic device to accomplish the emergency disconnect on command. At the present time, the emergency disconnect requirement has been replaced by a design philosophy of component redundancy to avoid operational difficulties. No investigations were initiated for the new requirement.

3.2.3.4 Vehicle Recovery Considerations

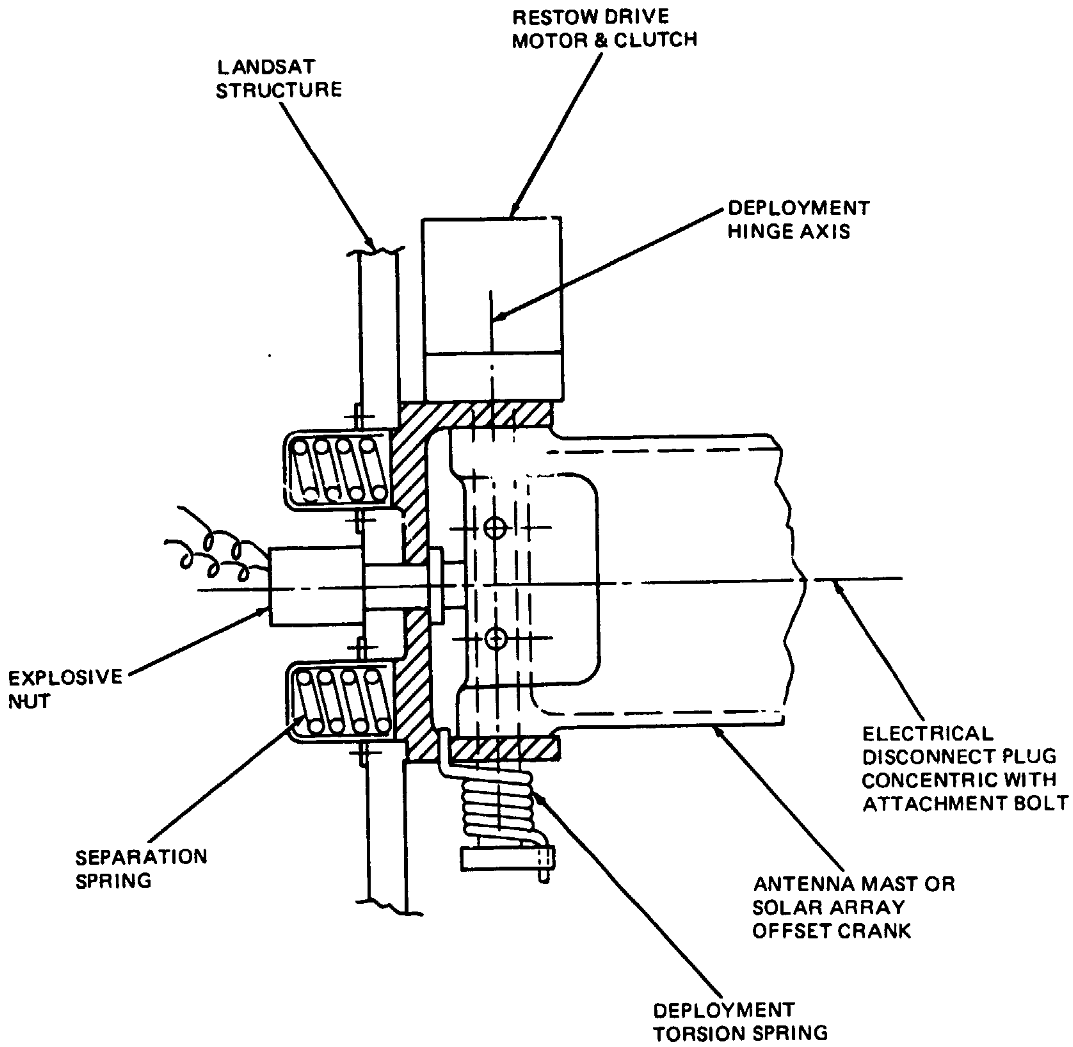
Landsat vehicle recovery by Shuttle operation is a program option for equipment service and/or maintenance. Due to the large volumetric capacity of the Shuttle equipment bay, it has been suggested that the TDRS antenna subsystem need not be restowed to the original launch configuration if such option offers design advantages. Pursuant to this suggestion, two antenna arrangements were reviewed for feasibility and design advantage. These two arrangements were:

- The antenna mast is left in the fully deployed position
- The antenna mast is partially retracted.

The first arrangement is not considered viable since the mast and its support structure will not survive re-entry loads without a significant increase in strength. It is questionable whether sufficient increase in strength can be achieved within size and weight limitations. Supports located in the Shuttle would be an added design complexity necessitating resolution in a number of interface areas.

The second arrangement is equally unattractive in that any partially retracted position will require additional locking mechanisms over and above that required for the initial launch configuration. Since partial retraction requires some form of drive mechanism, it is felt that it is just as easy to drive to a fully stowed position.

In summary, it is recommended that the TDRS antenna subsystem be driven to the fully stowed configuration for the Shuttle recovery operation.



2473-150

FIGURE 3.2-21 EMERGENCY DISCONNECT SYSTEM

3.3 STRUCTURAL ANALYSIS

The aims of the structural analysis are as follows:

- Determine the internal loads in the structure and hence size the structure so that a weight assessment can be made
- Determine the influence coefficients on the dynamic model so that the normal modes and frequencies can be calculated
- Determine the relative displacement of the structure per unit thermal gradient on the 'worst' face
- Analyze critical key structural members to demonstrate feasibility and provide backup to the weight assessment.

Trade studies to determine optimum material selection or an optimum structural configuration within the confines of the available envelope were not within the scope of the study.

The first three aims or goals are best achieved by generating a simple finite element model and by using readily available programs within the Grumman RAVES* and COMAP**-ASTRAL*** Systems. The fourth goal can only be achieved by the usual hand analysis.

The analysis has been confined to the T-1A configuration. This represents the more severe of the two cases as the Thematic Mapper has a weight of 650 lb for T-1A and only 250 lb for the H-1A. Since the configurations are similar, the structural deflections, natural frequencies, etc., obtained for the T-1A should be improved for the H-1A configuration.

Figure 3.3-1 shows the steps taken in the development of the structural model.

3.3.1 ASTRAL - Ideas Program S3

The purpose of the ASTRAL program is to provide the capability to analyze any arbitrary structure. The word analyze is taken in its broadest sense in that it

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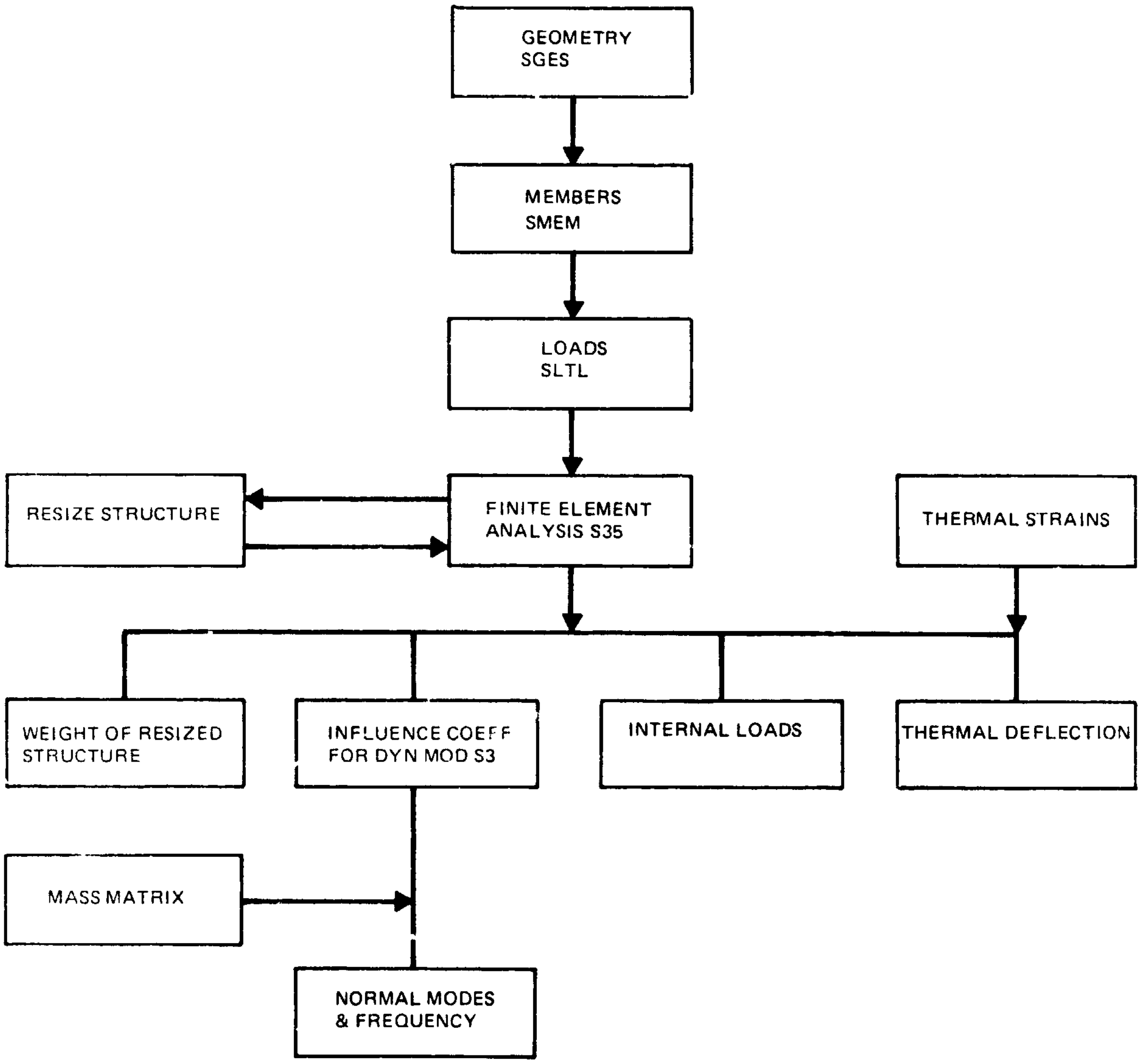
* Rapid Aerospace Vehicle Evaluation System

** Comprehensive Matrix Algebra Procedure

***Automated STRuctural AnaLysis



2473-119

FIGURE 3.3-1 STRUCTURAL ANALYSIS FLOW

means to compute internal member loads, deflection influence coefficients, etc. for a variety of imposed conditions such as external applied loads, thermal gradients, imposed displacements, induced strain, etc. This is a rather large requirement to force on a single program, especially if one considers that the structure must be arbitrary in nature. In order to accomplish this task, the ASTRAL program has been integrally built within the COMAP matrix package. The function of the ASTRAL system is to generate matrices that express the necessary relations between structural quantities using basic geometric, elastic, topological and boundary condition type data. These matrices are then operated upon by using the COMAP system to obtain the desired results. The system is thus completely flexible in that the engineer is allowed complete freedom in choosing a method of attack that best fits his particular needs.

3.3.1.1 Structures Model

The Structures Model conforms to the requirements of Grumman's ASTRAL finite element analysis procedure. The underlying principle behind the method is that every structure may be idealized into an assemblage of individual structural components or elements. The selection of elements for a particular aerospace component model is based on the similarity of their load-deformation characteristics to the actual structure.

The idealized elements are connected at discrete node points to which all loads are applied and transferred through the structure. Any restraints such as boundary conditions are also applied at these points.

The number of elements used to define the structure is based on past experience since undue refinement leads to a great increase in the amount of work required to analyze the structure and only complicates the interpretation of the results.

3.3.1.2 Program Description

ASTRAL utilizes the stiffness method to analyze redundant structures. Its library includes a wide variety of elements which can be connected to represent a given Structural Model.

The material properties of each element and its nodal relation to the entire structure, SMEM, are used in conjunction with the nodal geometry, CGES, to produce element stiffness matrices. These relate nodal forces to nodal displacements. In addition, several matrices are generated to describe other items such as internal element stresses or planar forces and shear flows as functions of nodal displacement. Considering the effect of boundary conditions, SECS, the element structure matrices are accumulated to form the stiffness of the entire structure. Based on a given set of loading conditions, SLTL, or any unit type of condition, corresponding deflections, reactions and internal forces can be obtained.

To obtain greater flexibility, ASTRAL has been incorporated into COMAP to form S3 - COMAP - ASTRAL. Within this framework the ASTRAL portion is used to generate the required component matrices while COMAP performs the matrix operations as well as the bookkeeping tasks. Control is exercised over the entire process since the coding of the required matrix operations is input with the data. In addition, several structural assemblies can be analyzed separately and then coupled in one computer submission.

3.3.2 T-1A Finite Element Model

The nodes of the structure are shown in Figures 3.3-2 through 3.3-4. Figure 3.3-2 illustrates the basic finite element model referenced to the YZ plane which is the interface between the Instrument Module and the MMS transition adapter. Figure 3.3-3 presents the nodes of the primary structure and Figure 3.3-4 shows the numbering system for the members. The following assumptions were made in developing the model:

- The model is held down rigidly at the nodes representing the interface with the adapter, i.e., nodes 1, 5, 6, 7, 11 & 12. This is a simplification, since the adapter is flexible, but is justified on the grounds of lack of information and time to take a more realistic approach.
- The instruments are supported by a tripod structure from the c.g. of the instrument to a hard point on the primary structure. In reality, the instruments are supported by special MEMs attachments. However, the sizing of the secondary structure is not within the scope of the model. Hence the model does not give the correct line of action at each instrument attachment

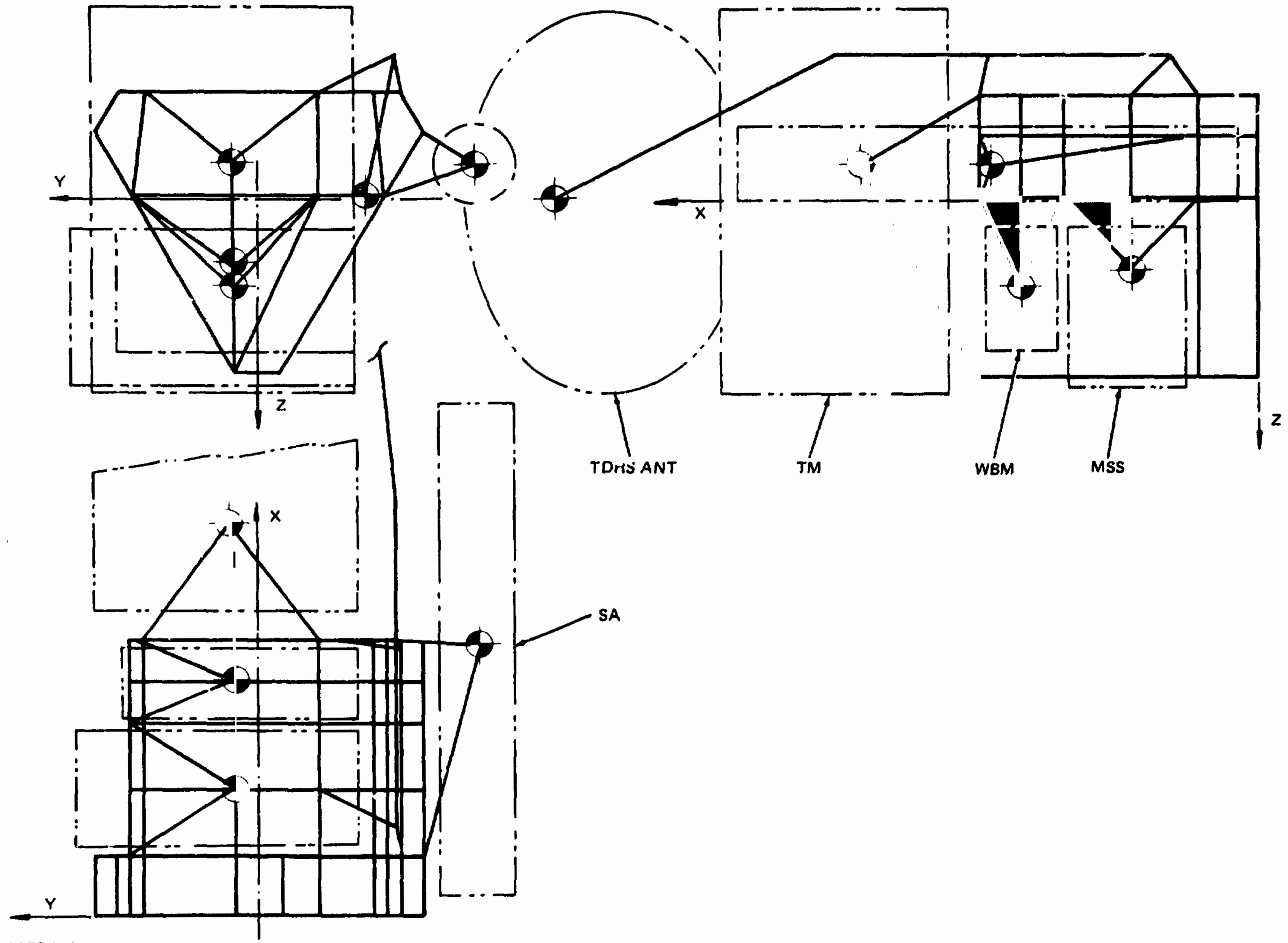


FIGURE 3.3-2 T-1A STRUCTURAL FINITE ELEMENT MODEL (SHEET 1 of 2)

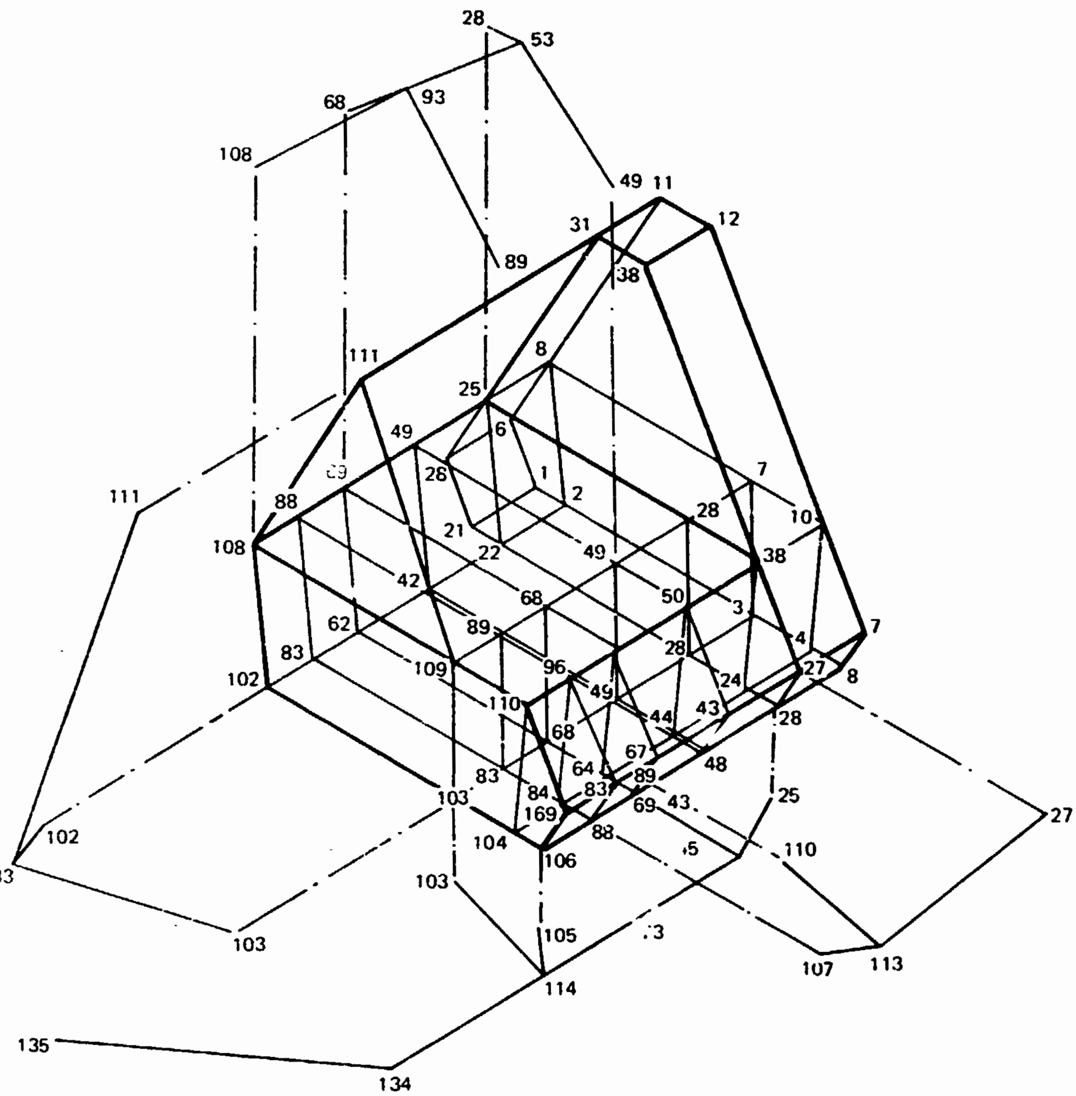


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FIGURE 3.3-2 T-1A STRUCTURAL FINITE ELEMENT MODEL (SHEET 2 of 2)

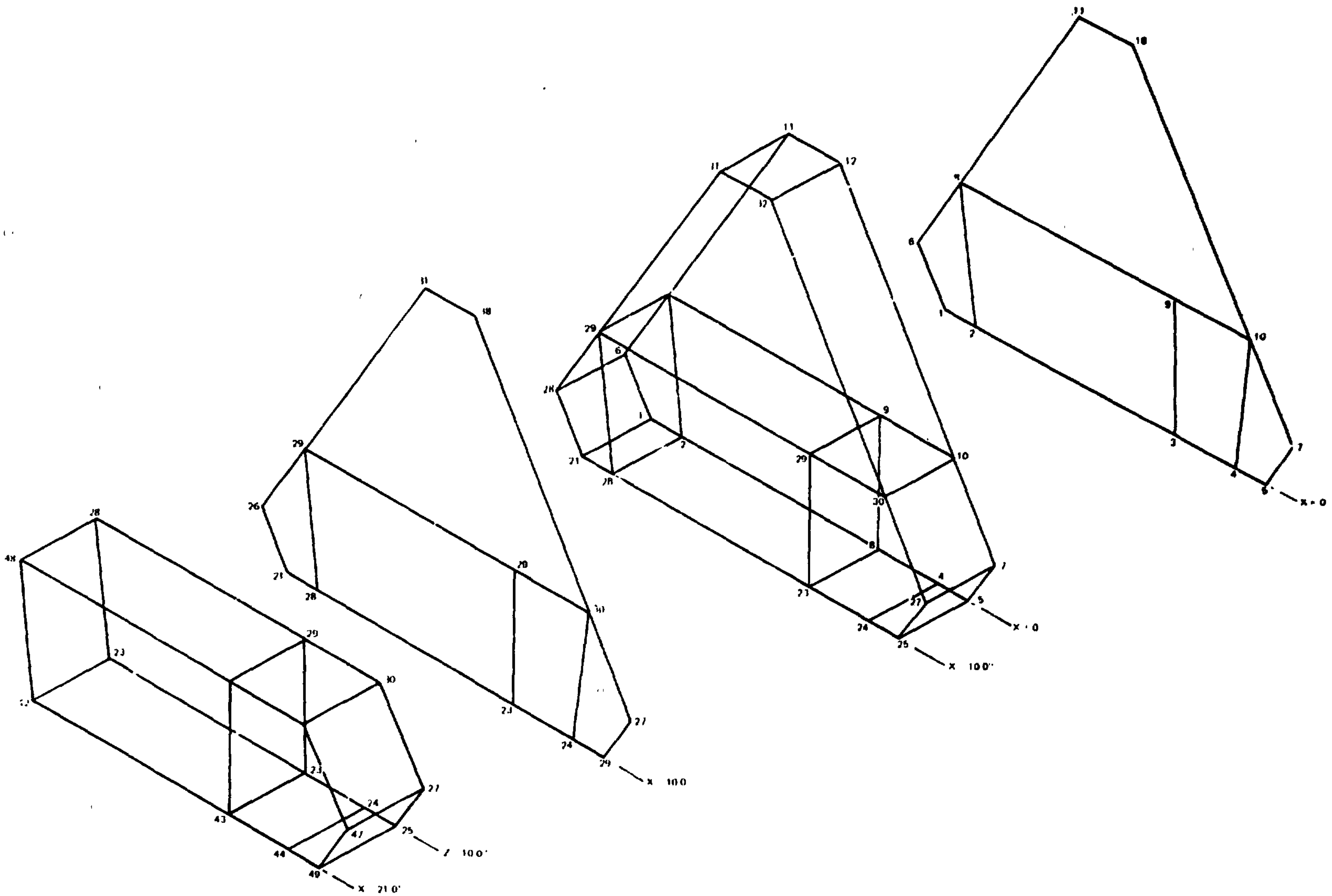

FIGURE 3.3-3 LANDSAT T-1A FINITE ELEMENT MODEL-PRIMARY STRUCTURE-MODES (SHEET 1 OF 2)

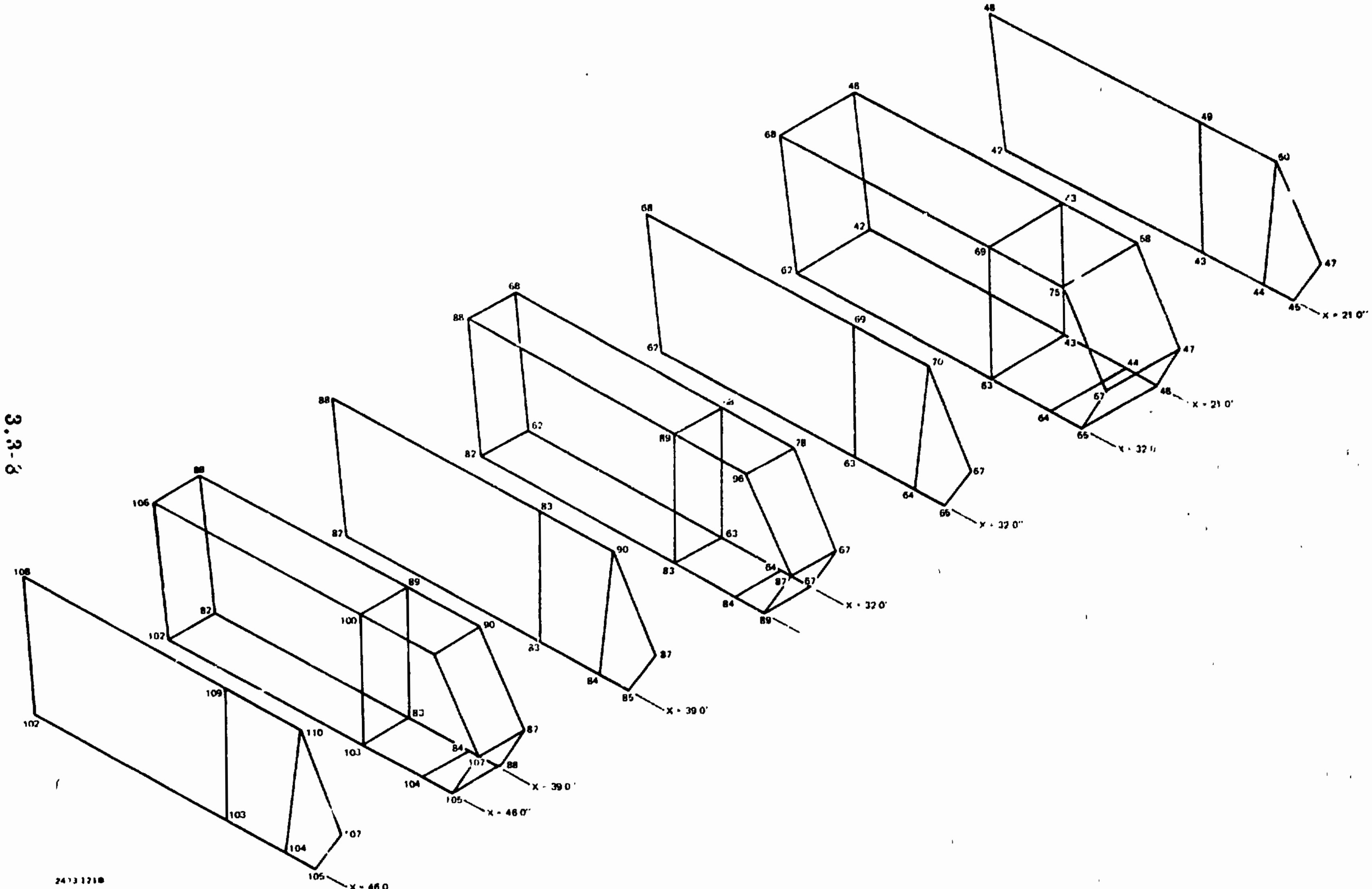

FIGURE 3.3-3 LANDSAT T-1A FINITE ELEMENT MODEL-PRIMARY STURCUTRE-MODES (SHEET 2 OF 2)

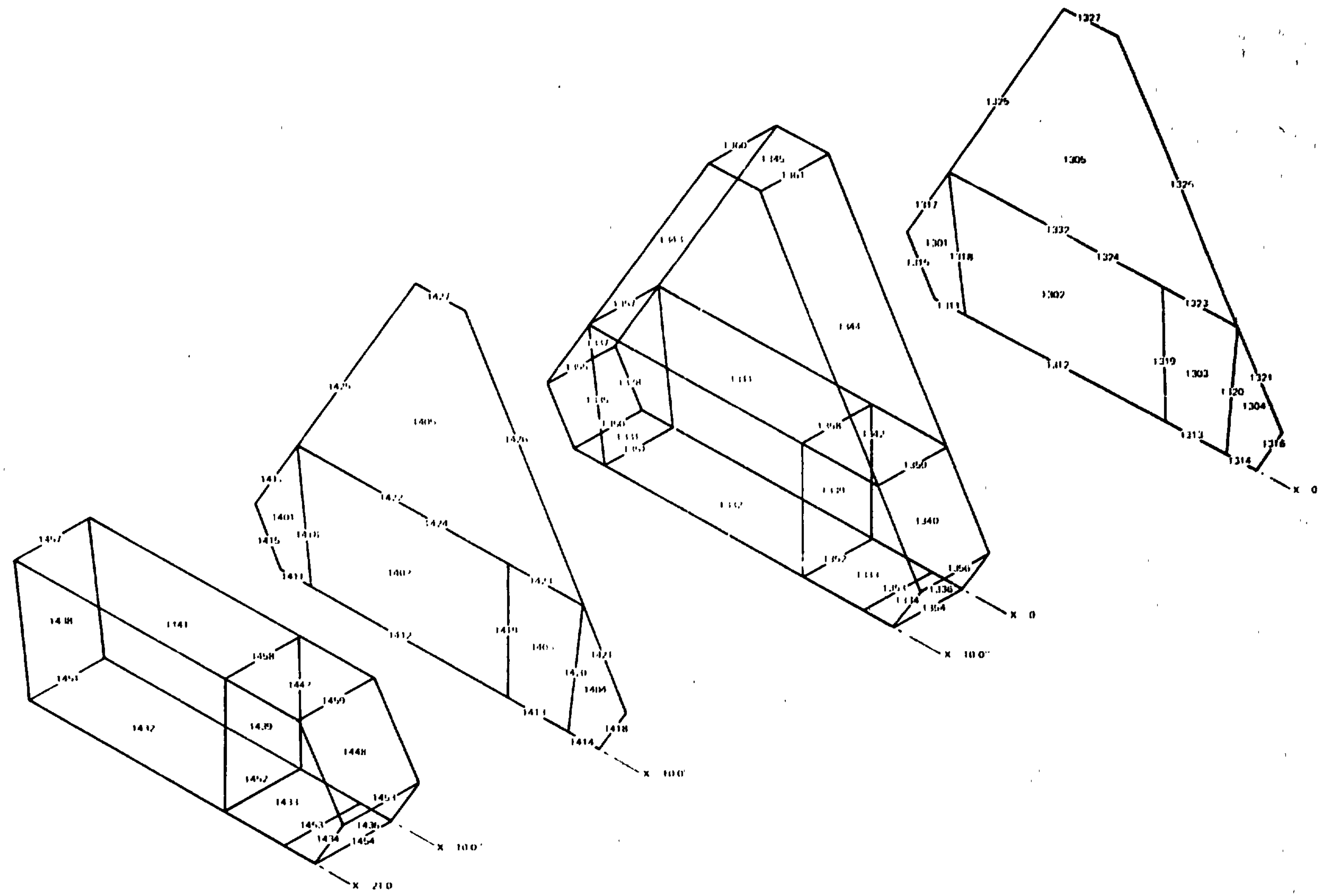

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FIGURE 3.3-4 LANDSAT T-1A FINITE ELEMENT MODEL PRIMARY STRUCTURE-MEMBERS (SHEET 1 OF 2)

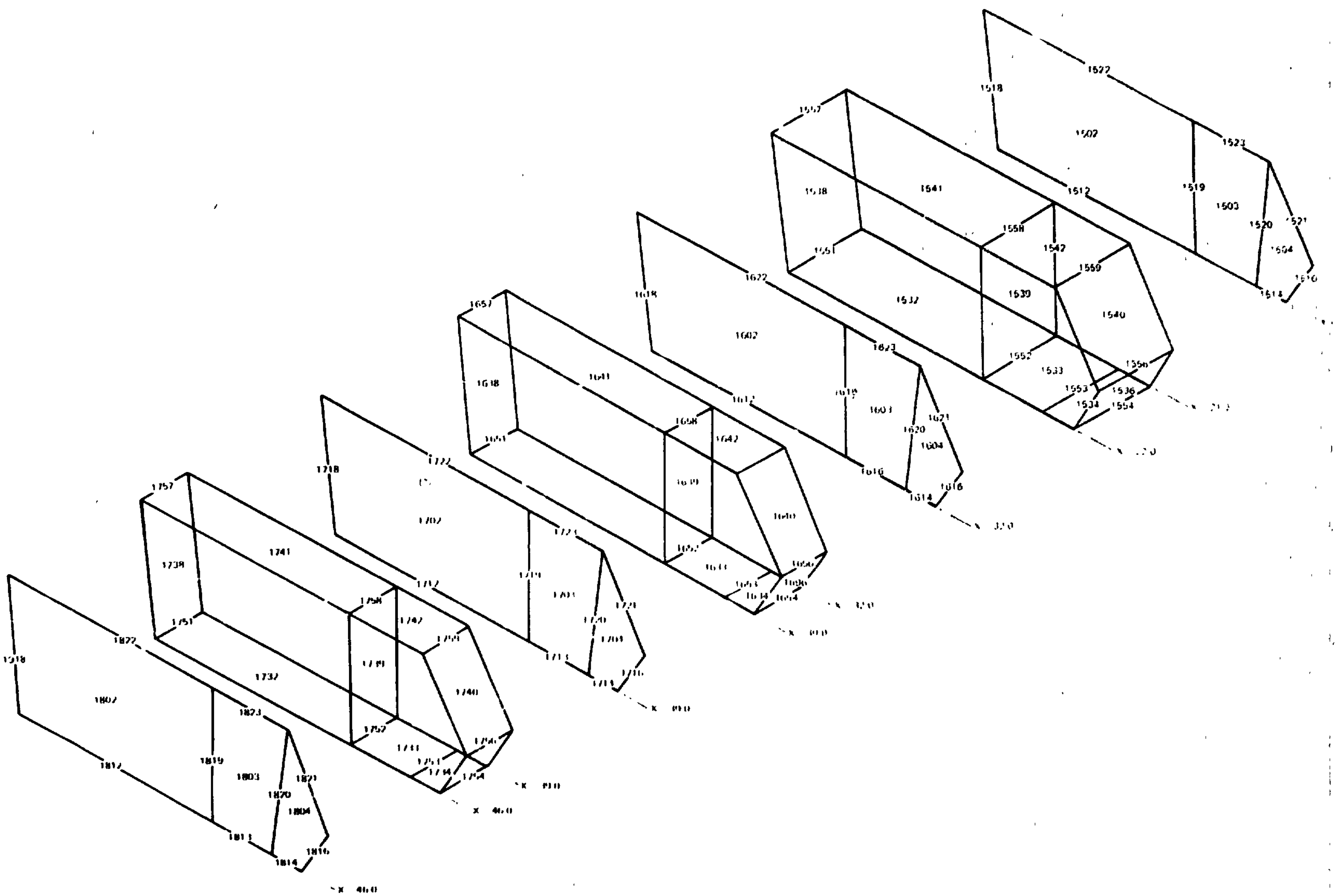

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FIGURE 3.3-4 LANDSAT T-1A FINITE ELEMENT MODEL PRIMARY STRUCTURE-MEMBERS (SHEET 2 OF 2)

point, but does have the virtue that the approximate flexibility of the attachment is represented.

The primary structure is modeled using Bar elements (element No. 1) and a Quadrilateral Shear Panel (element No. 6). The instruments are supported on bars with the exception of the TDRS antenna which is supported by a Beam element (element No. 2) in addition to bars.

The model has 135 modes, 216 members and 198 degrees-of-freedom.

3.3.3 External Loads

Inertia loads were applied to the model at 14 mass points, eight representing the structure and 6 representing the instruments, etc. (These mass points were subsequently used for the dynamic model). Table 3.3-1 gives the node number on the structural model, the description and mass.

The possible applied accelerations given in section 2.1 were reduced to give five loading conditions thought to give the critical cases for the structure. These are shown in Table 3.3-2. Due to the change in reference axis between the MMS and the Landsat, the accelerations must be resolved. This is illustrated in Figure 3.3-5.

3.3.4 Results

The significant results of the structural analysis are discussed in the following paragraphs.

3.3.4.1 Member Sizing

Having assembled the geometry, member and loads data for the structure, the internal loads for each member were calculated using RAVES program S35. The structure was then resized automatically using the same RAVES program. The members were coded and material cards supplied so that each member was sized using a stress level of 20 ksi for axially loaded members and 12 ksi for shear panels. The resulting bar areas and panel thicknesses are shown in Figure 3.3-6. Using the modified member data, the internal load distribution was rerun.

TABLE 3.3-1 MASS POINTS

Node No.	Item	Weight, Lb
22	Structure	67.37
23		91.90
28		158.70
29		216.48
35	MSS	148.0
73	TDRS Antenna Can	53.0
93	WBM	110.0
102	Structure	15.70
103		21.42
108		36.98
109		50.45
113	SA	150.00
133	TM	650.0
135	TDRS Antenna Hd	127.0

2473-161T

TABLE 3.3-2 ULTIMATE ACCELERATIONS, g's

Case No.	Condition	MMS Axes			Landsat Axes		
		X	Y	Z	X	Y	Z
1	MECO	-18.5	3.3	3.3	-18.5	1.21	4.51
2	MECO	-18.5	-3.3	3.3	-18.5	4.51	-1.21
3	Vibration	10.0	10.0	10.0	10.0	3.66	13.66
4	Vibration	10.0	-10.0	-10.0	10.0	-3.66	-13.66
5	Vibration	10.0	-10.0	10.0	10.0	13.66	-3.66

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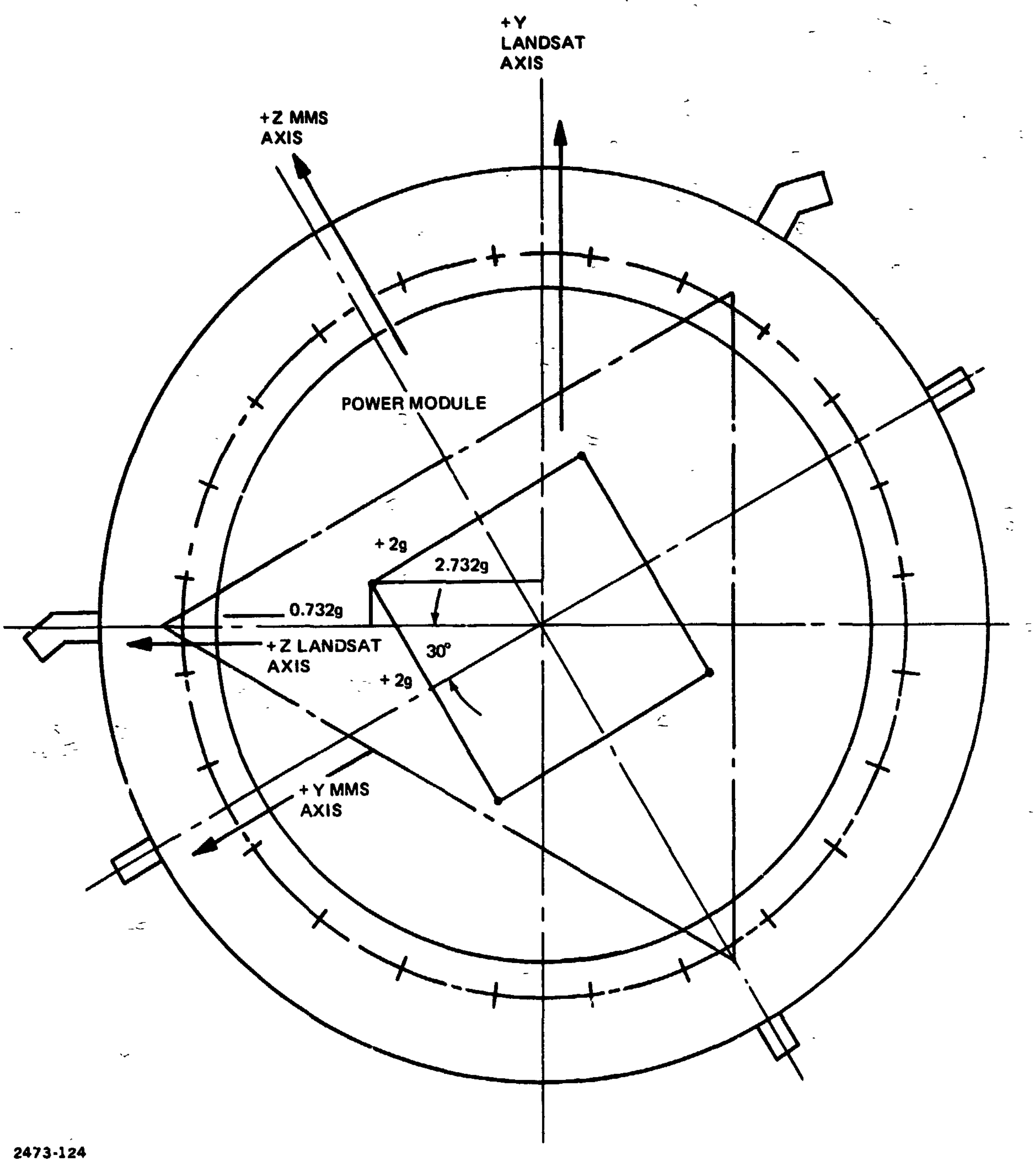

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FIGURE 3.3-5 T-1A REFERENCE AXIS–MMS AND LANDSAT

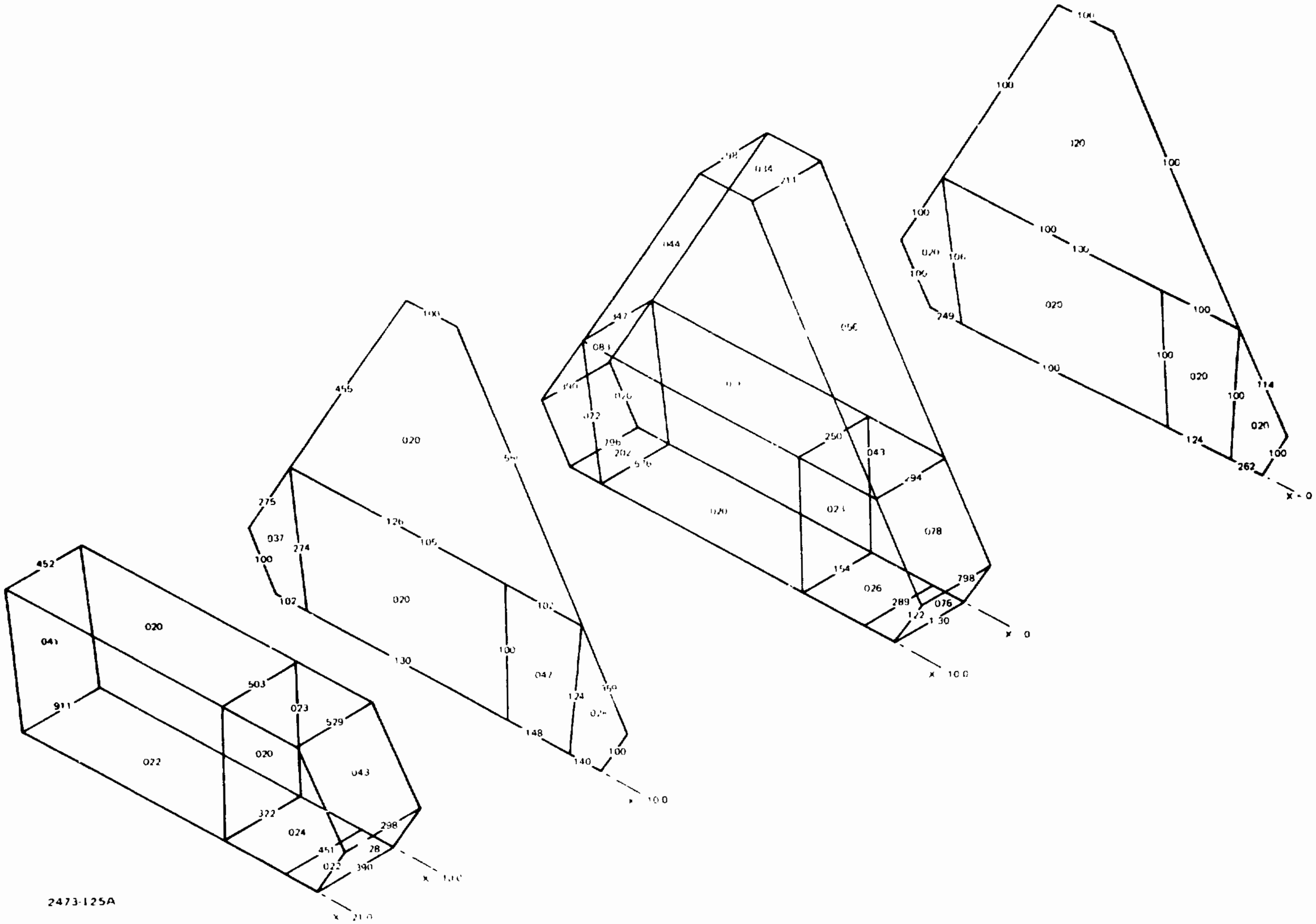

FIGURE 3.3-6 LANDSAT T-1A FINITE ELEMENT MODEL-RESIZED MEMBER DATA (SHEET 1 OF 2)

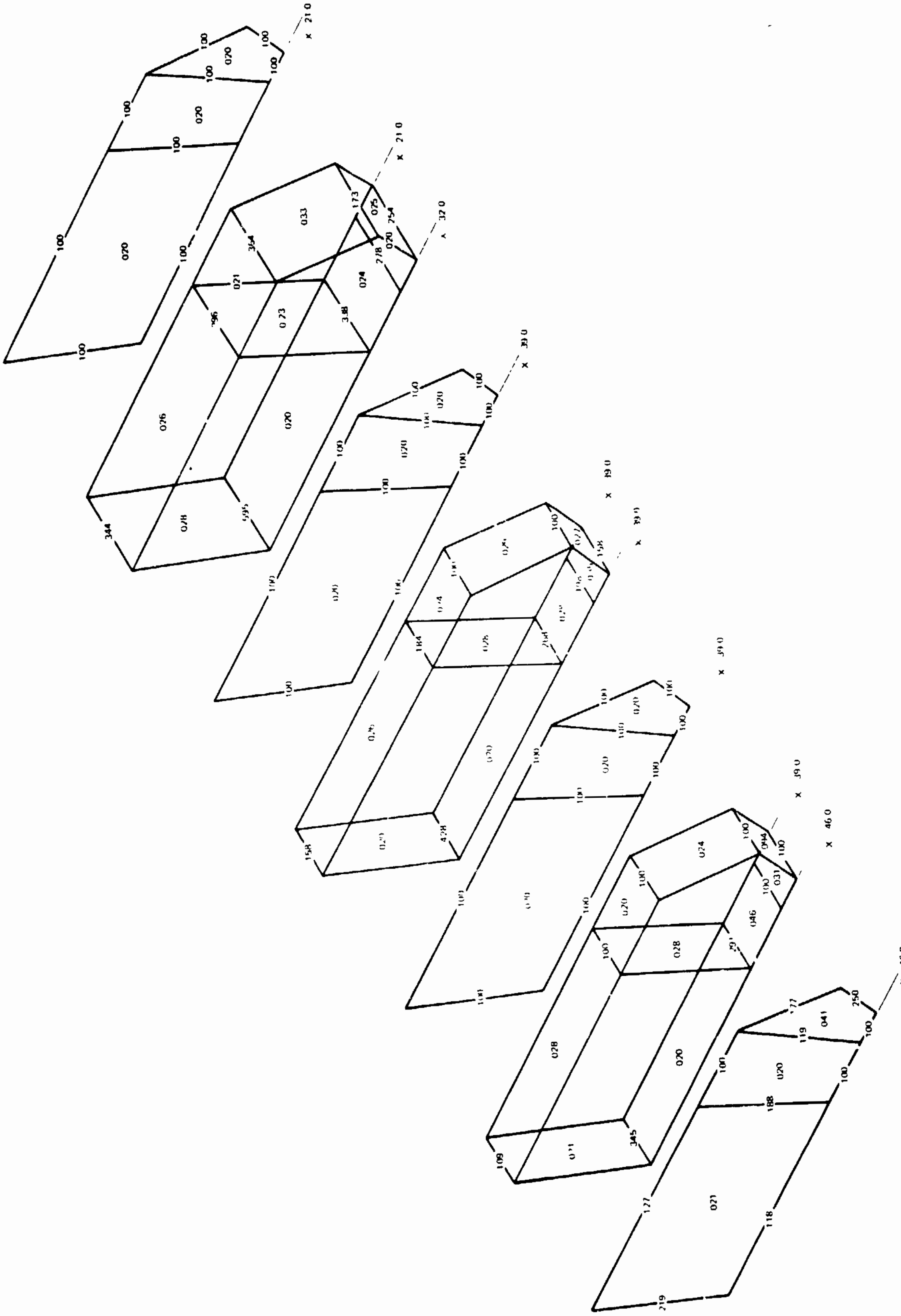

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FIGURE 3.3-6 LANDSAT T-1A FINITE ELEMENT MODEL-RESIZED MEMBER DATA (SHEET 2 OF 2)

As a typical example of the output of program S35, the idealization of the back face of the structure is shown in Figure 3.3-7. The maximum shear for the webs and the axial load for the bars is shown while the critical condition is noted in parenthesis. For example, shear panel 1633 has an ultimate shear of 399 lb/in. and the critical condition is 5. The allowable shear stress was put at 12 ksi, thus, the panel would be sized by the program at 399 ÷ 12000 = 0.033 in. Similarly, the bar load for member 1653 was 5023 and 3161 at each end. The allowable stress was set at 20 ksi and hence the program would set the area at 1/2 (5023 + 3161) ÷ 20000 = 0.20 in.^{2} Values shown in Figure 3.3-6 are slightly different because they represent the first iteration while these calculations represent the second iteration. However, the differences are very small.

In order to determine the optimum form of construction for the shear webs, a weight trade was undertaken. The configurations studies were: a stiffened sheet, a honeycomb panel, a corrugated sheet, and a tubular brace. The material in each case was aluminum alloy and it was assumed that each element should be shear resistant (non-buckling). The results are summarized in Figure 3.3-8. For the fully-webbed structures, the corrugations are shown to be the most efficient and the stiffened web the least efficient. The diagonal tube looks very favorable but further work is required to get a true comparison with the fully-webbed designs. This analysis justifies the allowable shear stress used in the computer resizing.

3.3.4.2 Displacement of Structure due to Unit Thermal Gradient

A gradient of 1°F on the -Z face of the structure gives 7.6 arc sec of rotation on the X = 46.0 face of the structure (between nodes 103 and 109).

3.3.5 Conclusions and Recommendations

The decision to use a finite element model at such an early stage in the development of the configuration has proven very beneficial. The major goals were achieved with accuracy and confidence. To accrue maximum advantage from this early start on the structural analysis using the model, the following steps are recommended:

(1) Modify the model to reflect and analyze possible solutions to the low frequency of the TDRS Antenna.

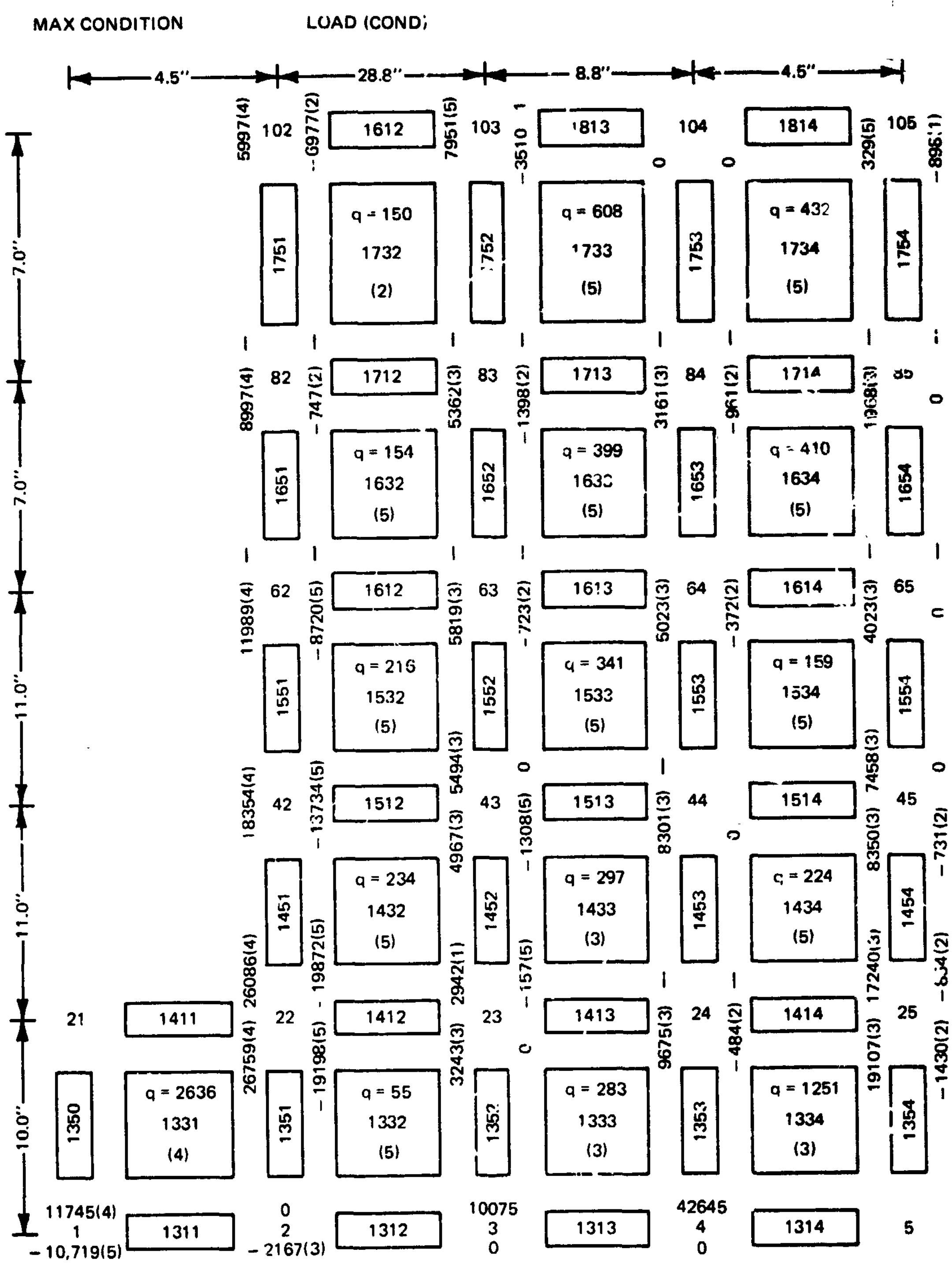

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FIGURE 3.3-7 TYPICAL OUTPUT OF S35 PROGRAM

(2) Modify the model to give rigid beaming of the instruments to flexible mountings on the structure. This requires a continuing effort as the MEMS systems develop.

(3) Elastically couple the model to the MMS (Multimission Modular Spacecraft). The assumption of a rigid attachment to the adapter gives rise to high interface loads due to short coupling between the fixed modes. Elastic coupling would have to be taken into consideration to determine the frequency of the complete spacecraft.

(4) Extend the thermal-structural analysis to support thermal design.

(5) Maintain the model to reflect configuration changes, e.g., member sizes and applied loads, as "feedback" occurs after this iteration.

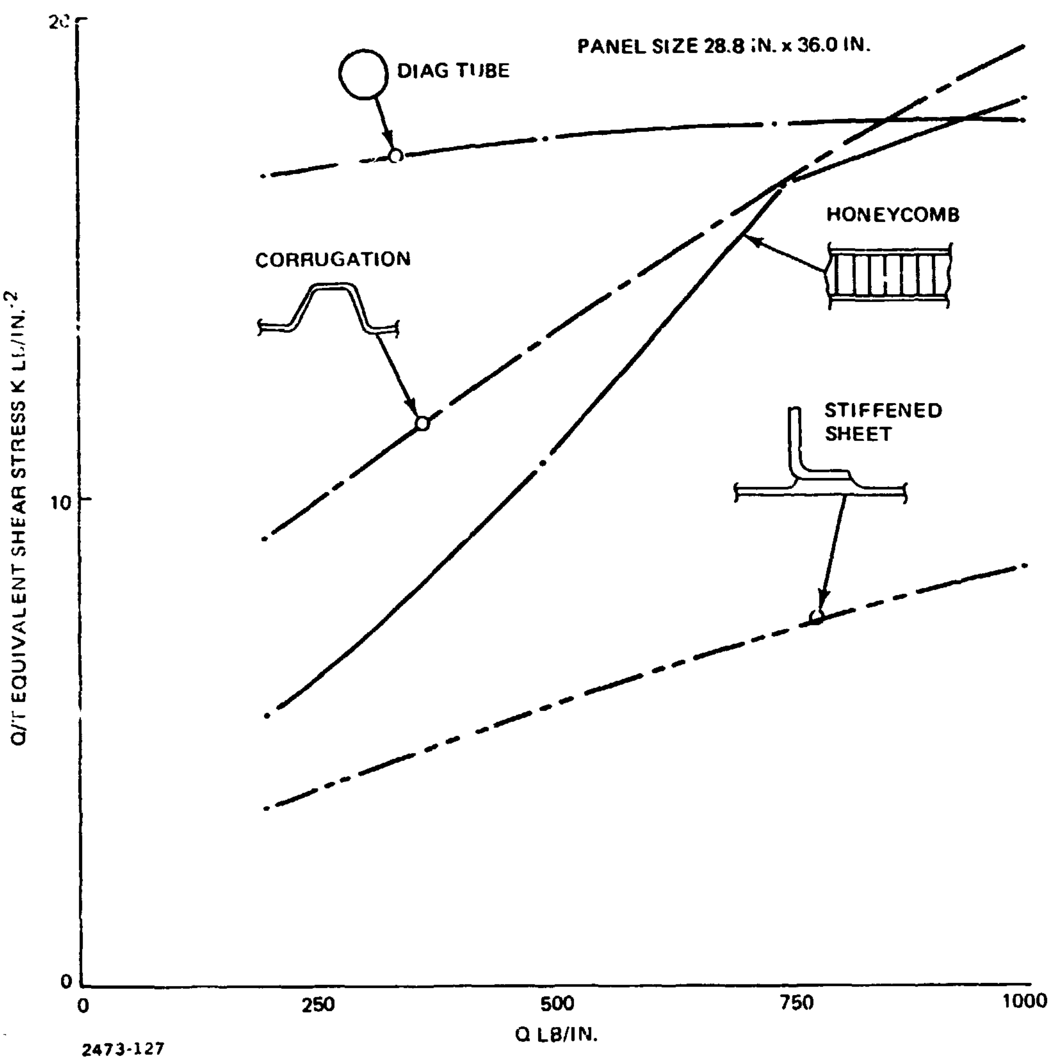

FIGURE 3.3-8 SHEAR WEB WEIGHT TRADE

3.4 DYNAMIC ANALYSIS

The dynamic analysis of the T-1A configuration Instrument Module consisted primarily of developing dynamic math models for the Launch and Boost Configuration and for the Orbital Configuration in order to determine structural modes and frequencies. This effort was preceded by the selection of Factors of Safety and the determination of design loads for primary structure and equipment support structure.

3.4.1 Factors of Safety and Preliminary Structural Design Load

The Factors of Safety are multiplicative constants applied to Limit Loads to obtain Design Loads. These factors were obtained from Table 2 of Reference 1 and are:

Ultimate Load = 1.65 x Limit Load

Qualification Test Load (Protoflight) = 1.5 x Limit Load

Preliminary Structural Loads for Primary Structure and Equipment Support Structure are shown in Table 3.4-1. Primary Structure Load Conditions I and II were obtained from Table 5 of Reference 1 and are a summary of the MMS Quasi-Static Acceleration Design Loads (Ultimate). Load Condition III is based on the maximum vibration response of the MMS Modules as specified on the footnote of Table 2-2 of Reference 2.

Equipment Support Structure Loads are based on an envelope of the maximum response of Lunar Module equipment plotted as a function of the equipment's weight. These levels were then adjusted to account for the difference in the acoustic environment between the Saturn and the Delta Launch Vehicles. These loads will be used to size the structure that ties the equipment to the Instrument Module Primary Structure.

3.4.2 Dynamic Math Models and Natural Frequencies

Two Dynamic Math Models of the T-1A Landsat Instrument Module were developed to determine the module's modes and frequencies for the Launch Boost Configuration and for the Orbital Configuration.

3.4.2.1 Launch and Boost Configuration

The Launch and Boost Configuration Dynamic Math Model is assumed to be cantilevered from the MMS Transition Adapter and is based on Structural Math Model utilizing 135 Structural Nodes (Refer to Subsection 3.3.2). Influence Coefficients at 14 Mass Points (8 for Primary Structure and 6 for equipment cg's) in the x, y and z direction (42 degrees-of-freedom) were obtained from this Structural Math Model. The weights, cg's and degrees-of-freedom for the 14 Mass Points are shown in Table 3.4-2. All weights used in this model are maximum allocated weights. The T-1A Launch and Boost Configuration Dynamic Math Model is shown in Figure 3.4-1. A summary of the first eleven flexible modes covering up to 60 Hz is shown in Table 3.4-3. The Mode Shapes of the first seven flexible modes are shown in Figures 3.4-2a through 3.4-2g. The first two modes (4.78 and 5.49 Hz) are TDRS Antenna modes due to the bending of the antenna boom. The frequencies of these two modes are unacceptably low since the associated stiffness produces excursions of over ±4 inches during Launch and Boost. The main reason the antenna has such low frequencies is its long boom. As a first cut, no attempt was made to tie the antenna boom to the primary structure to minimize its length. Looking at the T-1A Structural Arrangement, it appears that one of the antenna boom pickups could be rearranged so that it will reduce the boom's overhung length by approximately 30%. This change increases the antenna frequency by 70% and reduce its excursion by 66% to less than ±1.5 inches. The antenna frequencies could also be doubled by increasing the boom tube diameters by 60%. By combining portions of the above schemes, the antenna frequency could be increased to above 10 Hz which will reduce the antenna displacement to less than ±1.0 inch. This approach will be uitlized to update the Dynamic Math Model before it is used in the Forced Response Analyses.

As mentioned previously, the Math Model was based on the T-1A Structural Arrangement which carries the heavier TRW TM (650 lb vs 250 lb for the H-1A) and will have lower frequencies. The T-1A influence coefficients were also used to get an indication of the frequency changes if the Hughes TM were mounted on the T-1A structure. This was accomplished by changing Mass Point 13 from 650 lb to 250 lb. The results indicate that most frequencies, including the first two (antenna modes), remained unchanged with increased frequencies in the Pedestal

TABLE 3-4.1 STRUCTURAL DESIGN LOADS

- **Primary Structure**

Condition	Ultimate Load, g		
	x	y	z
I	±18.5	±3.3	± 0
II	±18.5	0	±3.3
III	±10.	±10.	±10.

- **Equipment Support Structure**

All Equipment Support Structure shall be designed to carry 9000 lb for a 100 lb item to 12,000 lb for a 600 lb item (linear interpolation for other weights). These loads are to be applied for 10,000 cycles.

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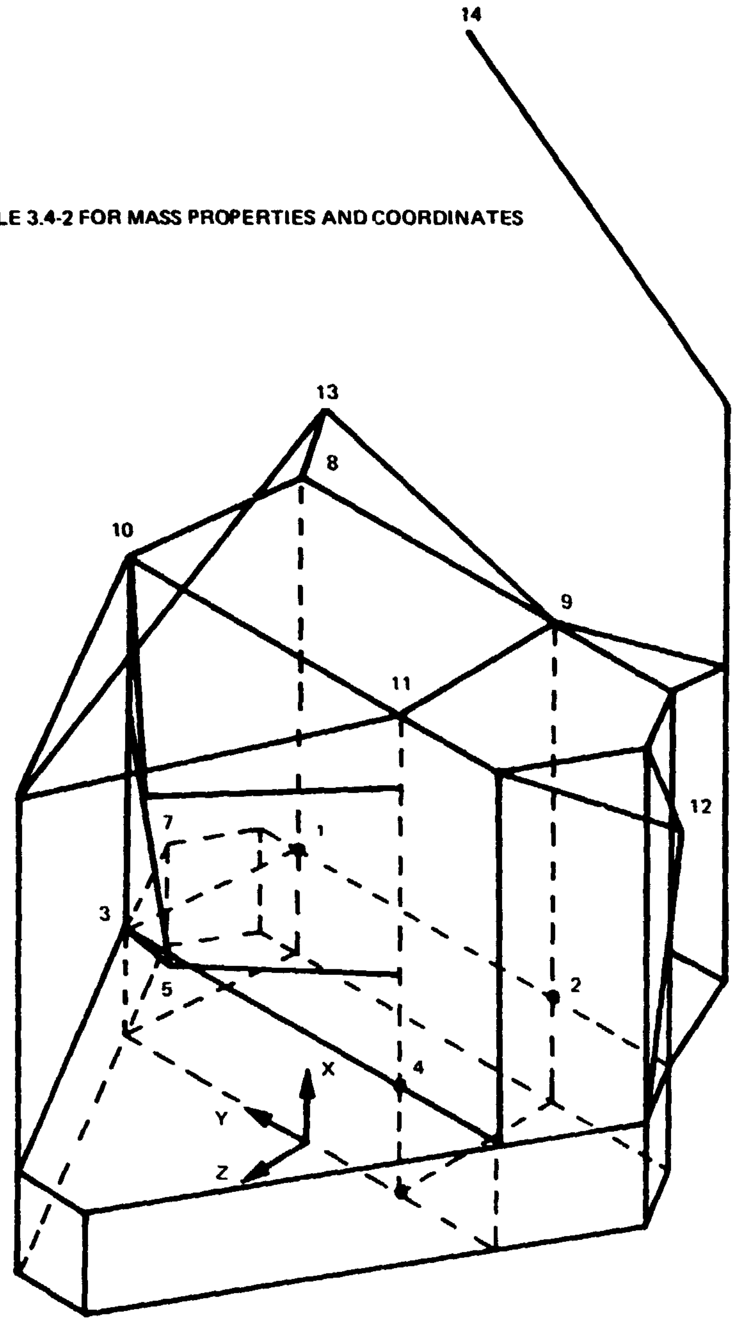
TABLE 3.4-2 LANDSAT T-1A INSTRUMENT MODULE LAUNCH AND BOOST DYNAMIC MATH MODEL

Mass Point	Description	Wt. Lb.	Coordinates			-of-Freedom		
			x	y	z	..	y	z
1	Structure	67.37	10	18.8	−17.8	1	2	3
2	Structure	91.9	10	−10	−17.8	4	5	6
3	Structure	158.7	10	20.9	−0.5	7	8	9
4	Structure	216.48	10	−10	−0.5	10	11	12
5	Multi-Spectral Scanner (MSS)	148	21.0	4	11.5	13	14	15
6	Antenna Can	53	29.5	−22.5	−24.5	16	17	18
7	Wide Band Module (WBM)	110	39.0	4	14.5	19	20	21
8	Structure	15.7	46	18.8	−17.8	22	23	24
9	Structure	21.42	46	−10	−17.8	25	26	27
10	Structure	36.98	46	20.9	−0.5	28	29	30
11	Structure	50.54	46	−10	−0.5	31	32	33
12	Solar Array	150	45.0	−36.0	−6.0	34	35	36
13	Thematic Mapper	650	65.5	4.4	−5.9	37	38	39
14	TDRS Antenna	127	116.0	−17.5	0	40	41	42
ΣWt. = 1897 lbs.								

2473-108T

NOTE: SEE TABLE 3.4-2 FOR MASS PROPERTIES AND COORDINATES



2473-109

FIGURE 3.4-1 T-1A LAUNCH AND BOOST DYNAMIC MATH MODEL

TABLE 3.4-3 LANDSAT T-1A INSTRUMENT MODULE LAUNCH AND BOOST CANTILEVERED MODES

Mode	Frequency, (Hz)	Generalized Mass, (Lb)	Description
1	4.78	136	Antenna Dish Lat Translation (Y)
2	5.49	162	Antenna Dish Lat Translation (Z)
3	14.4	1071	Pedestal Lat Bending (Z)
4	22.5	799	Pedestal Lat Bending (Y)
5	29.9	322	Pedestal Torsion (OX)
6	32.5	198	WBM Vert Translation (X)
7	40.7	769	TM Vert Translation (X)
8	44.1	285	MSS Vert Translation (X)
9	52.3	267	Antenna Dish Vert Translation (X)
10	53.5	862	MSS & WBM Lat Translation (Y & Z)
11	60.0	688	SA Vert and Lat Translation (X & Y)

Note: First seven mode shapes are shown in Figures 3.4-2a through -2g

2473-110T

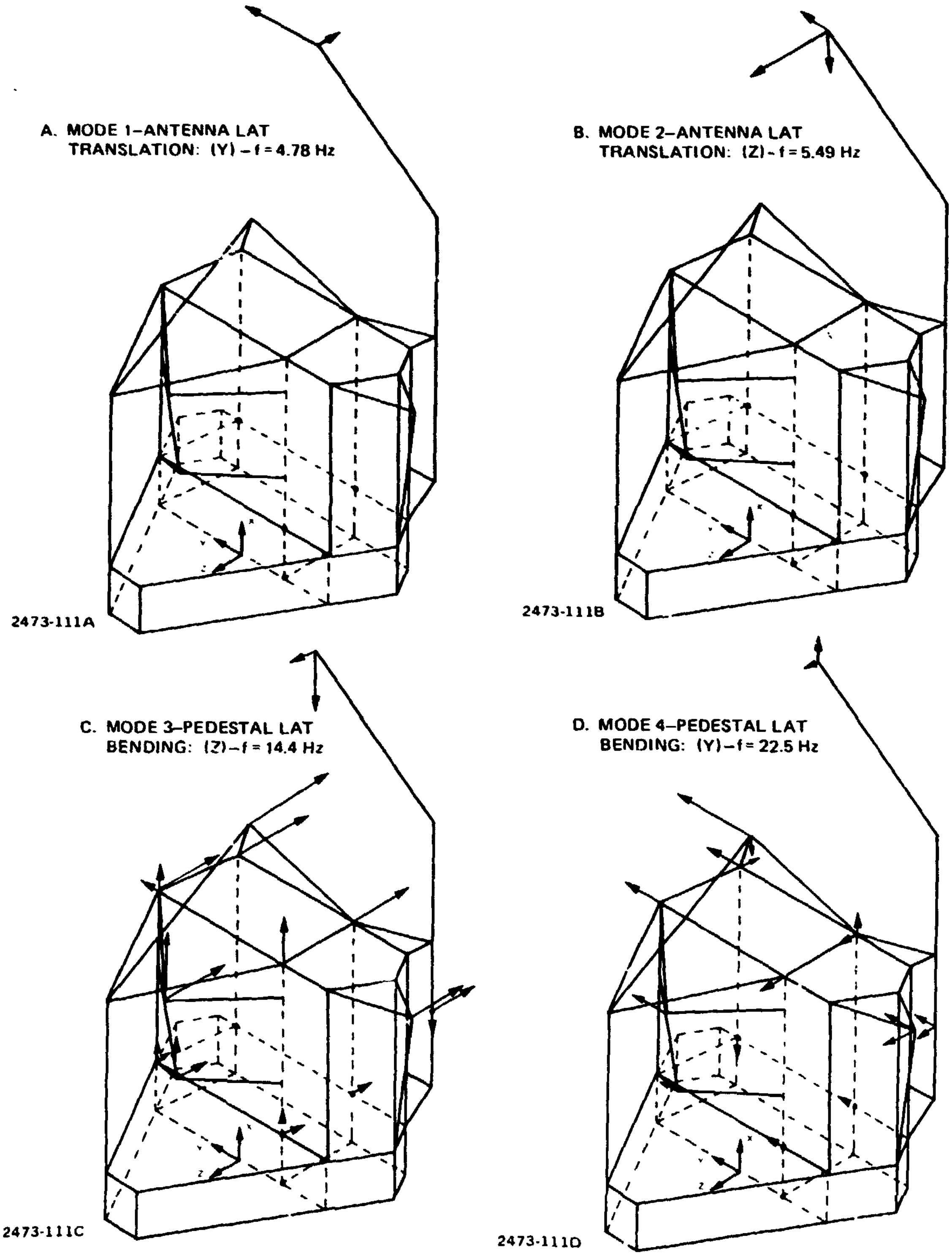


FIGURE 3.4-2 T-1A LAUNCH AND BOOST CANTILEVERED MODE SHAPES (SH 1 of 2)

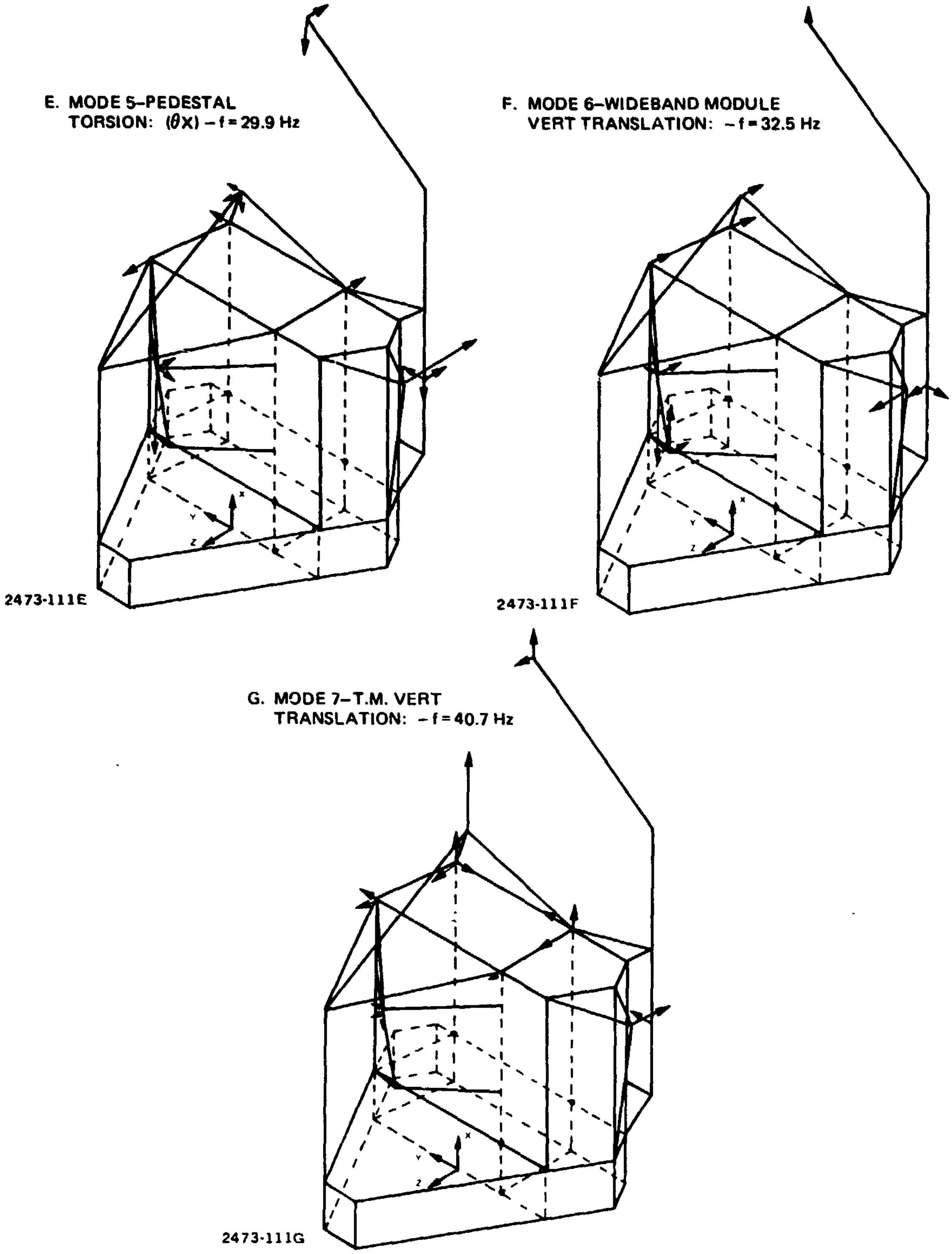


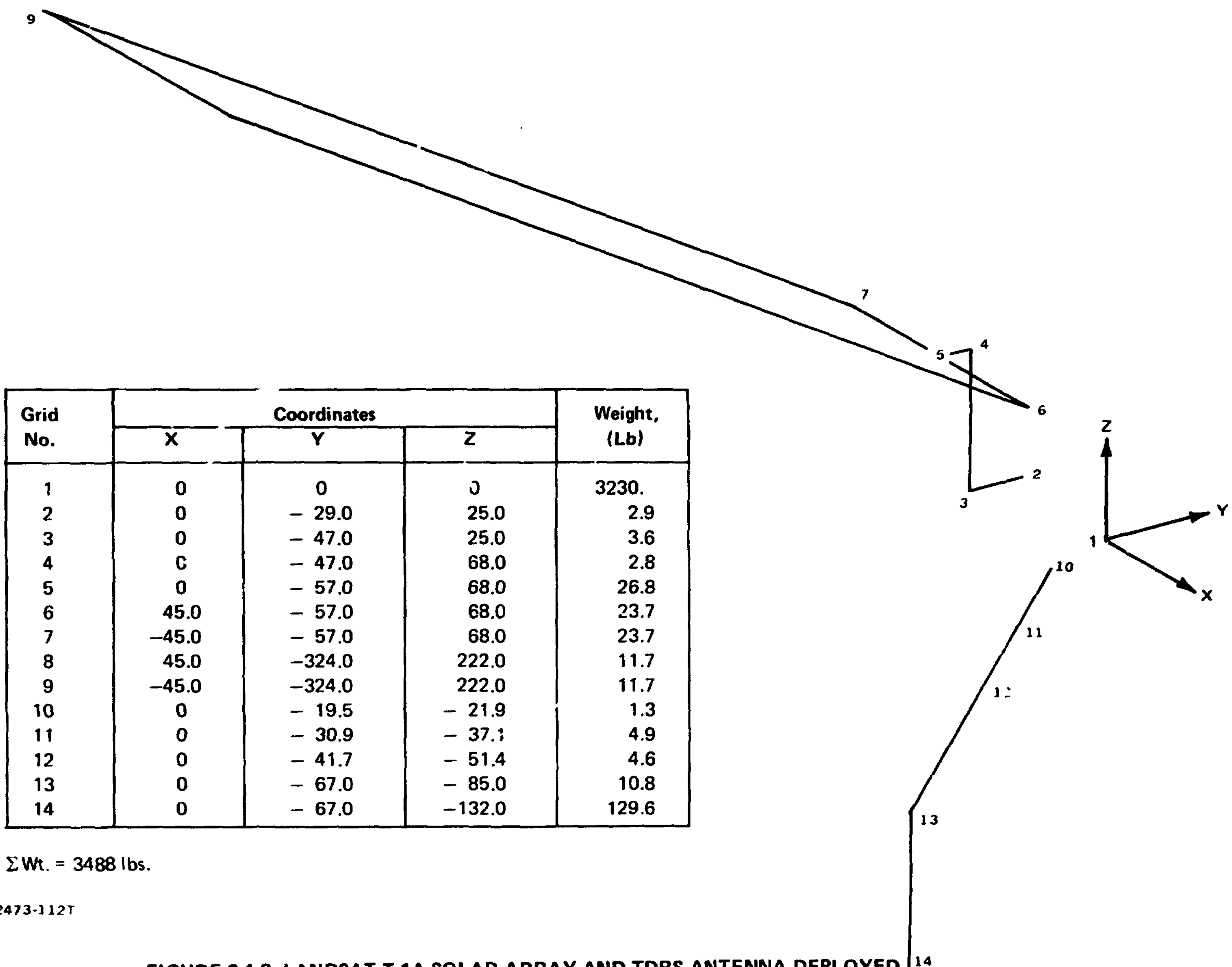
FIGURE 3.4-2 T-1A LAUNCH AND BOOST CANTILEVERED MODES SHAPES (SH 2 of 2)

Bending Modes (14.4 to 18.0 Hz and 22.5 to 35.1 Hz) and in the TM Vertical Translation Mode (40.7 to 54.1 Hz). Again it should be pointed out that these frequencies are valid only if the T-1A structure is identical to the H-1A structure. However, since the two structures and the equipment arrangement are not the same, the results are only indicative.

3.4.2.2 Orbital Configuration

The T-1A Landsat Spacecraft (Instrument Module, MMS and Adapter) was also analyzed in its Orbital Configuration. In this configuration, the extended Solar Array and TDRS Antenna were expected to have very low frequencies starting around 0.3 Hz. Based on the Launch and Boost Vibration Analysis, the Instrument Module's frequencies other than the appendages were expected to be above 15Hz. Therefore, the Landsat Spacecraft On-Orbit vibration analysis is based on a rigid spacecraft with a flexible Solar Array and TDRS Antenna. The analysis was performed using NASTRAN. The Dynamic Math Model shown in Table 3.4-3 and corresponding NASTRAN data described in Table 3.4-4, was represented by 14 grid points (9 for the Solar Array and 5 for the Antenna) and 11 Bar Members (7 for the Solar Array and 4 for the Antenna). Each grid point was allowed 6 degrees-of-freedom so that the Solar Array and the TDRS Antenna is an 84 degree-of-freedom Dynamic Math Model. Unlike the Launch and Boost Dynamic Math Model whose mass properties were based on allocated weights, the Solar Array and the Antenna boom in this model used estimated weights based on the generalized masses shown in Table 3.4-5. The solar cells were assumed to have a density of 0.2 lb per sq ft. Fourteen modes (6 rigid body modes and 8 flexible modes) were analyzed and covered up to 10.8 Hz. A summary of the first eight flexible modes are shown in Figures 3.4-4a through 3.4-4h.

Harmonic Forced Response on the T-1A Landsat Spacecraft In-Orbit Math Model to a unit torque (1 in-lb) about the Spacecraft's x, y and z axis was also performed using a damping value g = 0.01. Responses (g) in the x, y and z direction due to unit torques about the x, y and z axis on the Solar Array (grid point 8) are shown in Figures 3.4-5a through 3.4-5j. Similar responses (g) on the Antenna (grid point 14) are shown in Figures 3.4-6a through 3.4-6i. These curves could be linearly scaled



Grid No.	Coordinates X	Coordinates Y	Coordinates Z	Weight, (Lb)
1	0	0	0	3230.
2	0	– 29.0	25.0	2.9
3	0	– 47.0	25.0	3.6
4	0	– 47.0	68.0	2.8
5	0	– 57.0	68.0	26.8
6	45.0	– 57.0	68.0	23.7
7	–45.0	– 57.0	68.0	23.7
8	45.0	–324.0	222.0	11.7
9	–45.0	–324.0	222.0	11.7
10	0	– 19.5	– 21.9	1.3
11	0	– 30.9	– 37.1	4.9
12	0	– 41.7	– 51.4	4.6
13	0	– 67.0	– 85.0	10.8
14	0	– 67.0	–132.0	129.6

ΣWt. = 3488 lbs.

2473-112T

FIGURE 3.4-3 LANDSAT T-1A SOLAR ARRAY AND TDRS ANTENNA DEPLOYED FREE-FREE DYNAMIC MATH MODEL-RIGID SPACECRAFT

TABLE 3.4-4 SOLAR ARRAY AND TDRS ANTENNA NASTRAN DATA

Bar No.	Description: Name	Description: Material, In.	Length, In.	$I_{yy}=I_{zz}$ ($In.^4$)	Grid No. That Bars Connect
1	Solar Array Offset Crank	3.0 D, 0.04 In. AL	18	0.424	2 & 3
2	Solar Array Offset Crank	3.0 D, 0.04 In. AL	43	0.424	3 & 4
3	Solar Array Cannister	8.0 D, 0.06 In. AL	45	12.06	5 & 6
4	Solar Array Cannister	8.0 D, 0.06 In. AL	45	12.06	% & 7
5	Solar Array Extender Tube	2.5 D, 0.005 In. STL	320	0.0307	6 & 8
6	Solar Array Extender Tube	2.5 D, 0.005 In. STL	320	0.0307	7 & 9
7	Solar Array Spreader Bar	2.0 D, 0.05 In. AL	90	0.157	8 & 9
8	Antenna Boom	7.0 D, 0.06 In. AL	19	8.08	10 & 11
9	Antenna Boom	8.0 D, 0.06 In. AL	18	12.06	11 & 12
10	Antenna Boom	8.0 D, 0.06 In. AL	42	12.06	12 & 13
11	Antenna Dish	6.0 D, 0.06 In. AL	47	5.09	13 & 14

Notes:

(1) Rigid Bars connect grid 1 to 2 and 10 and grid 4 to 5.

(2) Torsional stiffness, J, equal 2 Iyy except for bars 5 and 6 which have no torsional stiffness; values are assumed to be zero.

(3) Bending and Torsional Moduli Are

	Aluminum	Steel
E	10.5×10^6	29.5×10^6
G	4.0×10^6	11.0×10^6

2473-116T

TABLE 3.4-5 SOLAR ARRAY AND TDRS ANTENNA DEPLOYED FREE-FREE MODES

Mode	Frequency, Hz	Gen Mass, Lb	Description
1	0.328	52.8	Solar Array 1st Bend in YZ Plane
2	0.455	38.1	Solar Array 1st Bend in X Plane
3	0.487	31.4	Solar Array 1st Torsion (0_y & 0_z)
4	2.24	84.2	Offset Crank Bending in X Axis
5	2.83	70.7	Offset Crank Torsion (0_z)
6	3.11	116.	Offset Crank Bending in Y Axis
7	4.09	248	Antenna Bending in X Axis
8	5.73	392	Antenna Bending in Y Axis

Note:

(1) Mode shapes are shown in Figures 3.4-4A through -4H.

2473-117T

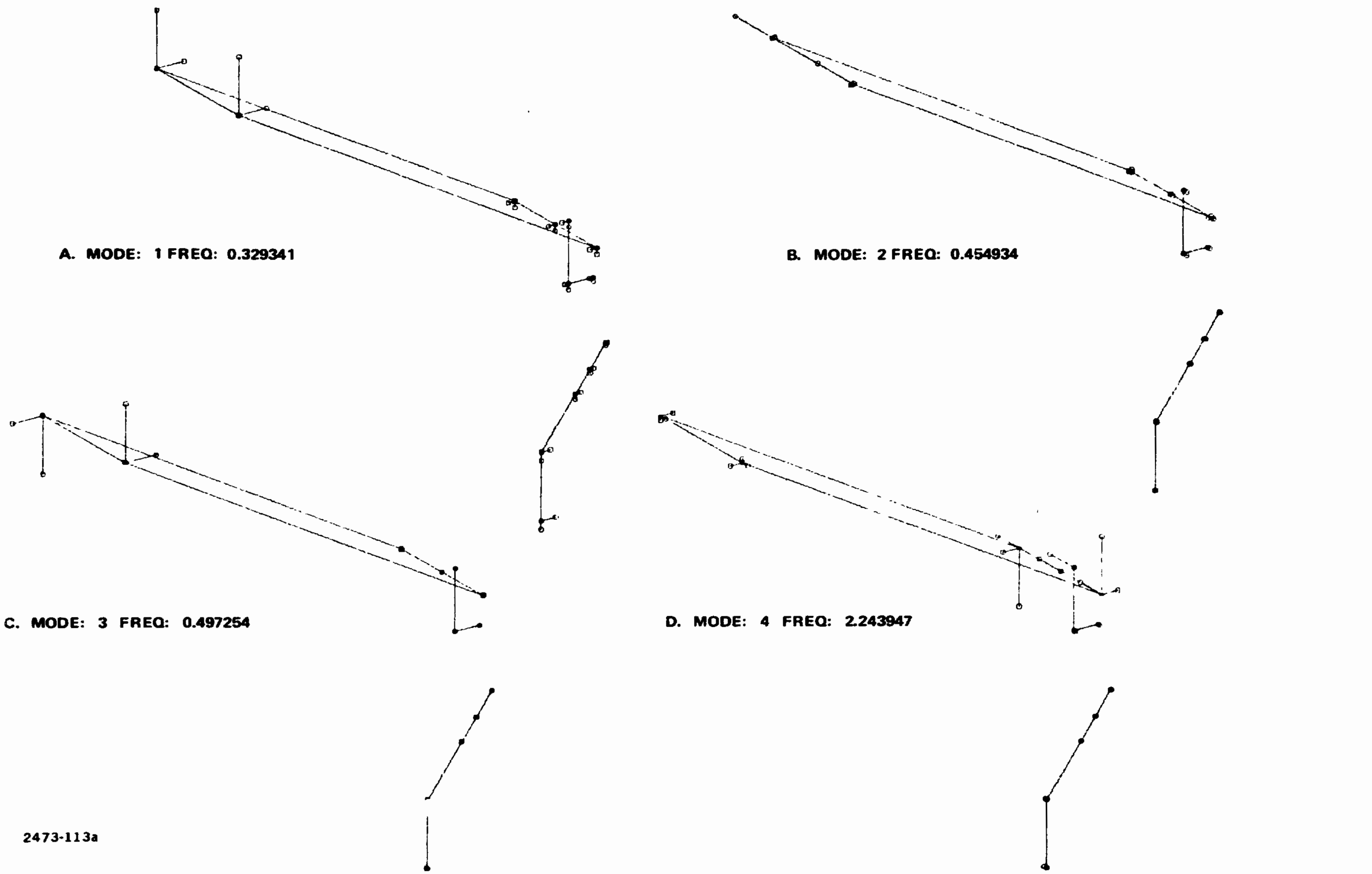


FIGURE 3.4-4 SOLAR ARRAY AND TDRS ANTENNA MODE SHAPES (SHEET 1 OF 2)

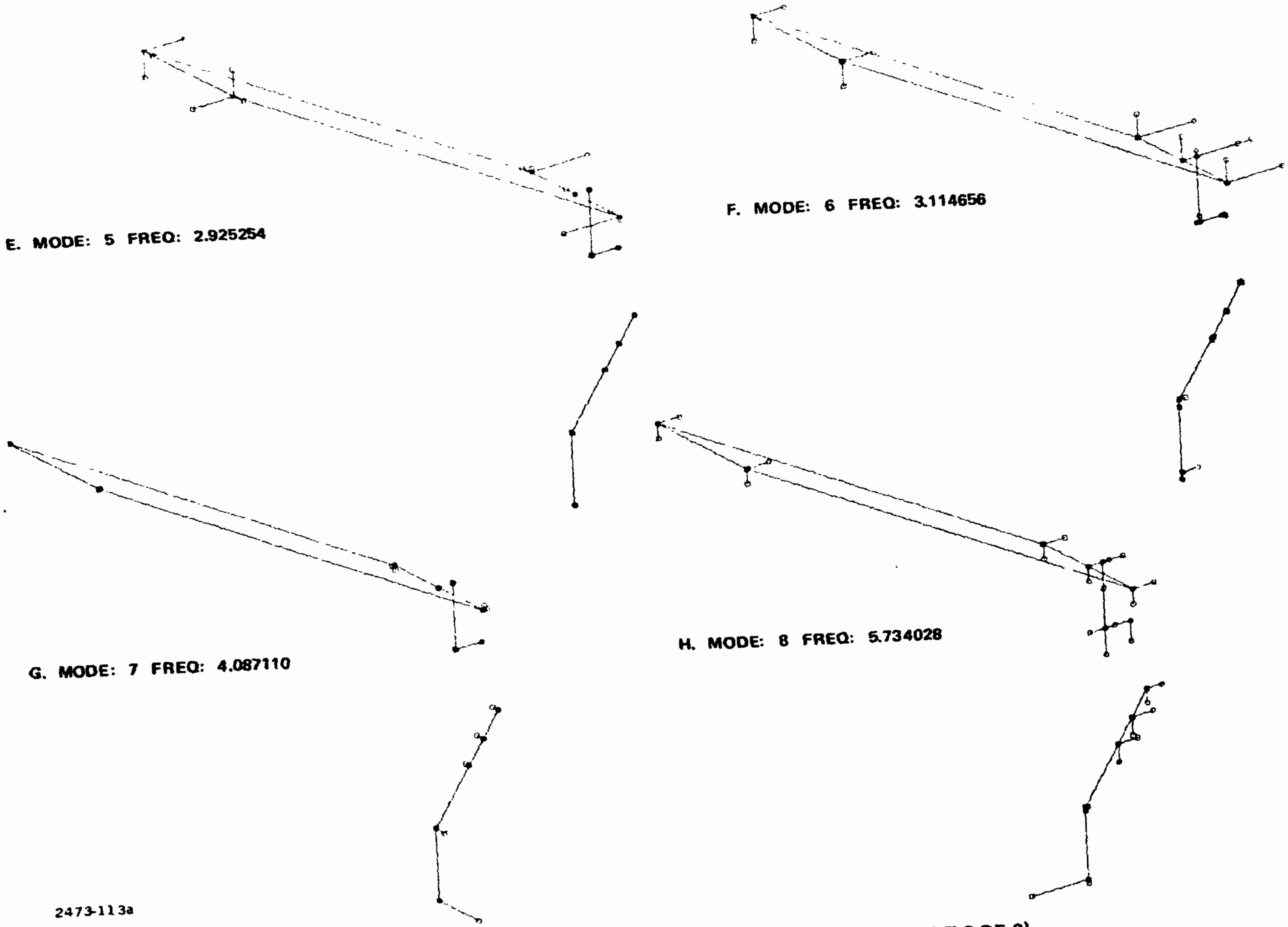
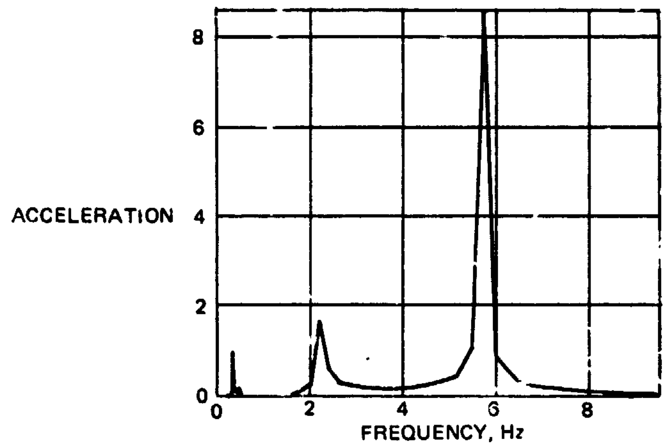
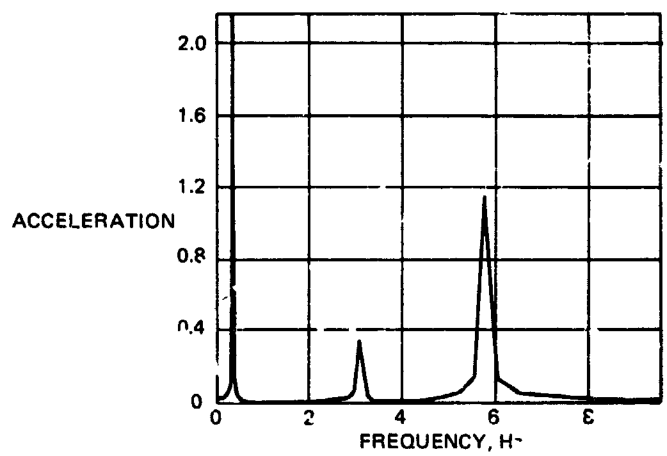


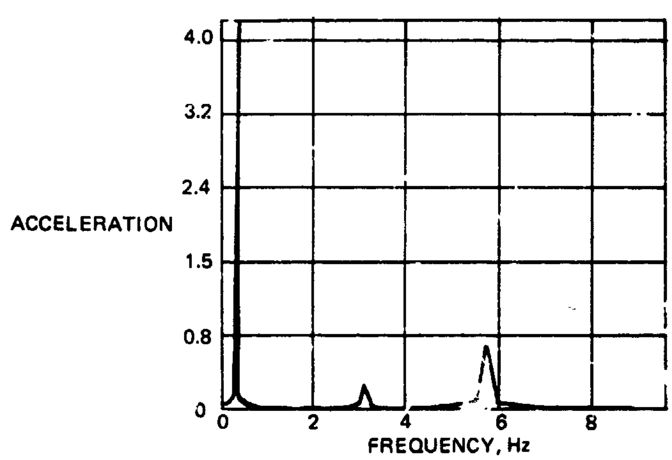
FIGURE 3.4-4 SOLAR ARRAY AND TDRS ANTENNA MODE SHAPES (SHEET 2 OF 2)



2473-114A A. X DIRECTION
UNIT TORQUE ABOUT SPACECRAFT X-AXIS

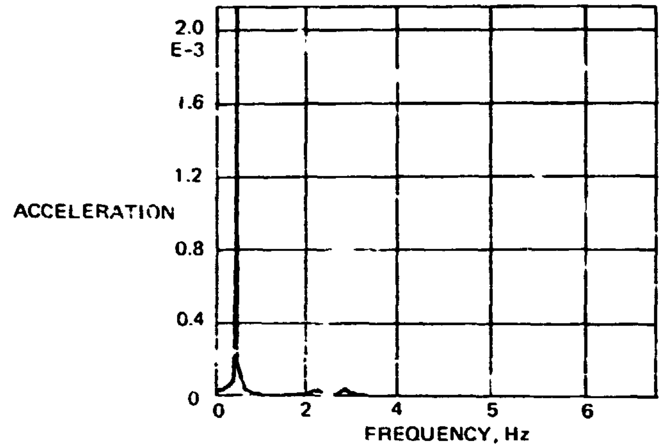


2473-114B B. Y DIRECTION
UNIT TORQUE ABOUT SPACECRAFT X-AXIS

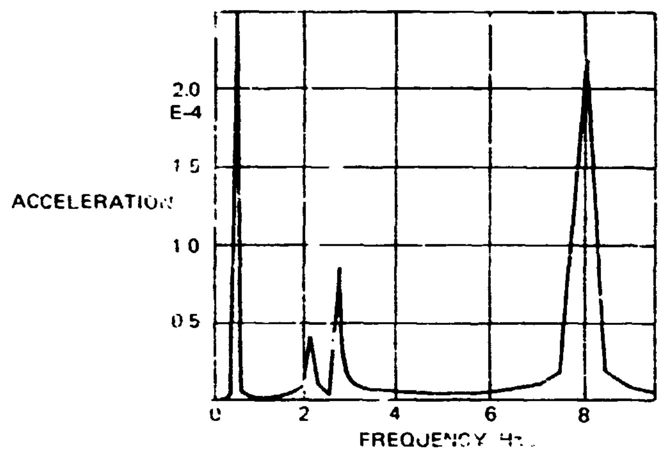


2473-114C C. Z DIRECTION
UNIT TORQUE ABOUT SPACECRAFT X-AXIS

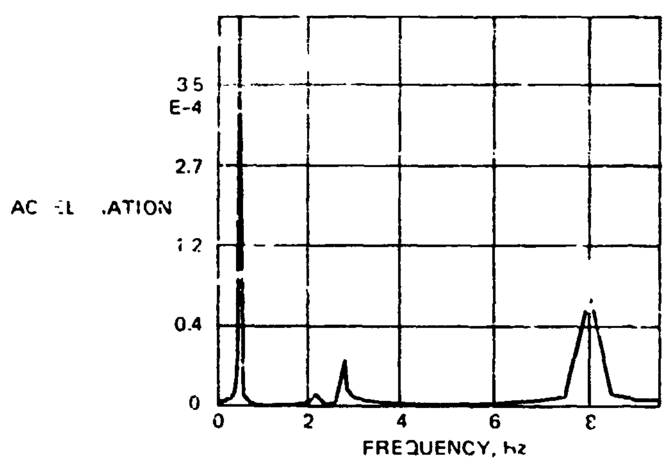
FIGURE 3.4-5 T-1A CONFIGURATION HARMONIC RESPONSE OF SOLAR ARRAY (SHEET 1 OF 3)



2473-114D D. X DIRECTION
UNIT TORQUE ABOUT SPACECRAFT Y-AXIS

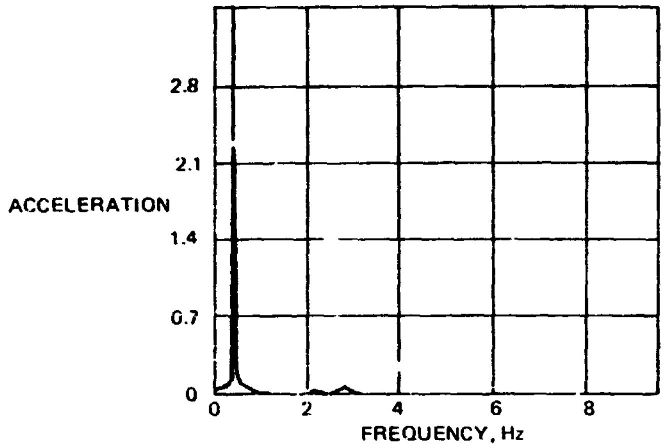


2473-114E E. Y DIRECTION
UNIT TORQUE ABOUT SPACECRAFT Y-AXIS

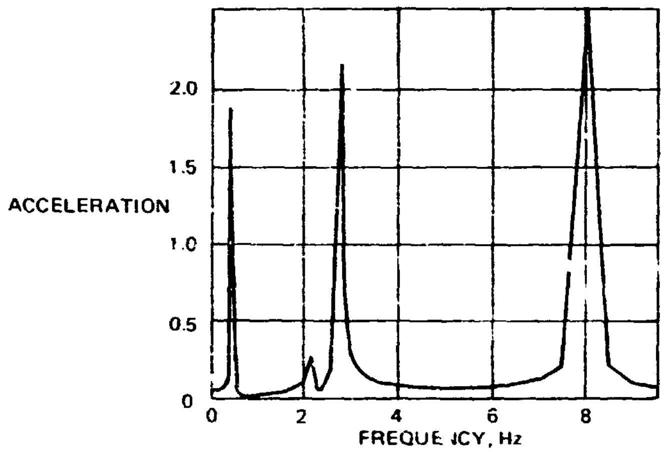


2473-114F F. Z DIRECTION
UNIT TORQUE AB... SPACECRAFT Z-AXIS

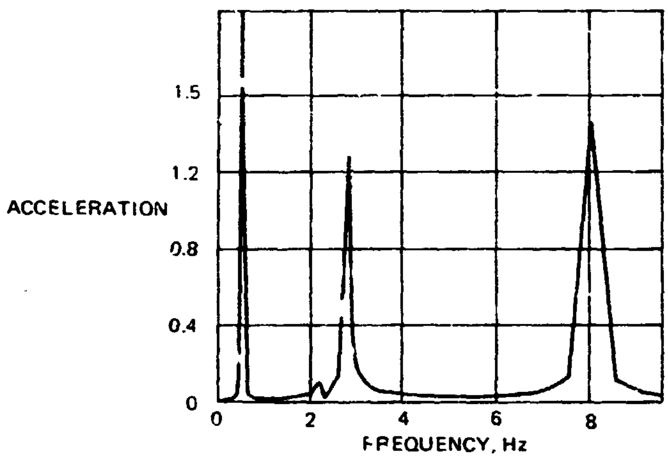
FIGURE 3.3-L T-1A CONFIGURATION HARMONIC RESPONSE OF SOLAR ARRAY (SHEET 2 OF 3)



2473-114G G. X DIRECTION
UNIT TORQUE ABOUT SPACECRAFT Z-AXIS

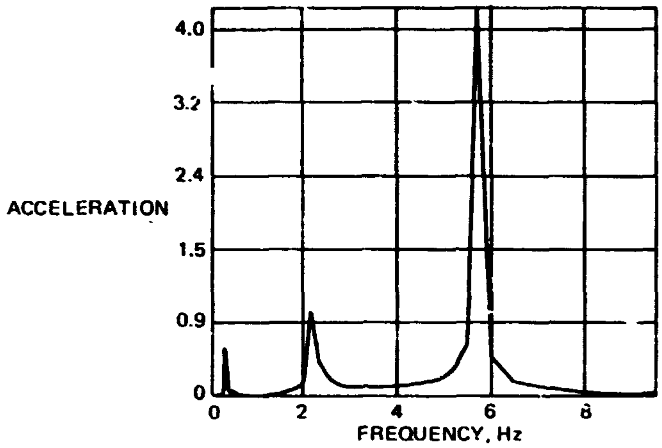


2473-114H H. Y DIRECTION
UNIT TORQUE ABOUT SPACECRAFT Z-AXIS



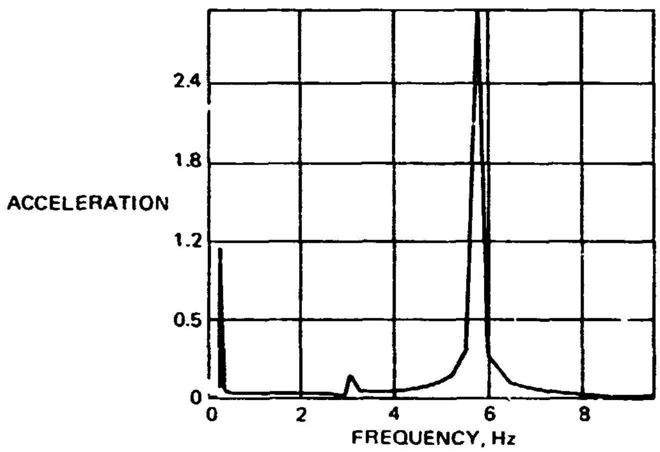
2473-114I I. Z DIRECTION
UNIT TORQUE ABOUT SPACECRAFT Z-AXIS

FIGURE 3.3-5 T-1A CONFIGURATION HARMONIC RESPONSE OF SOLAR ARRAY (SHEET 3 OF 3)



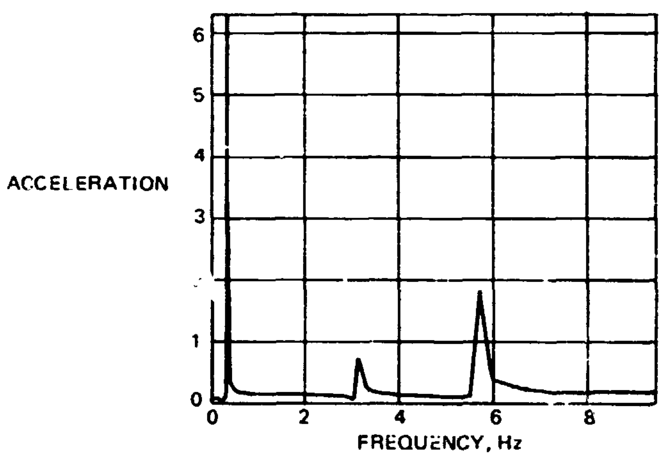
A. X DIRECTION
UNIT TORQUE ABOUT SPACECRAFT X-AXIS

2473-11A



B. Y DIRECTION
UNIT TORQUE ABOUT SPACECRAFT X-AXIS

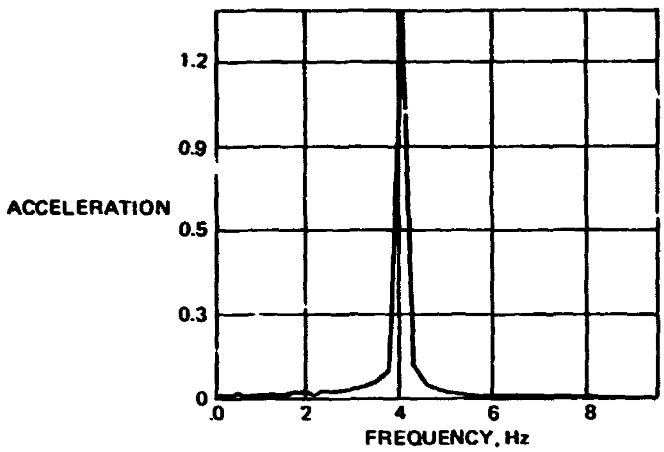
2473-115B



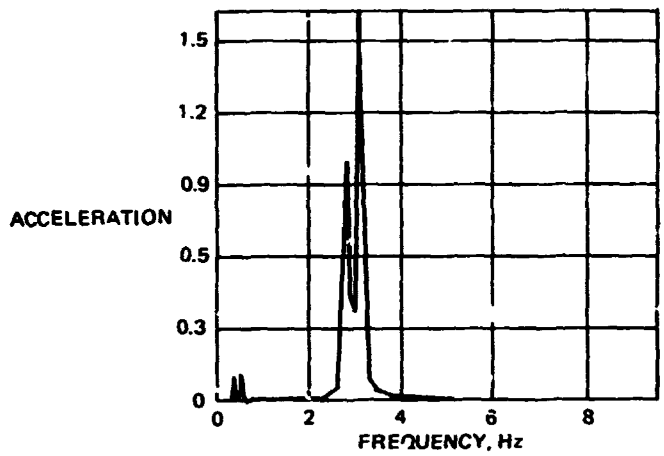
C. Z DIRECTION
UNIT TORQUE ABOUT SPACECRAFT X-AXIS

2473-115C

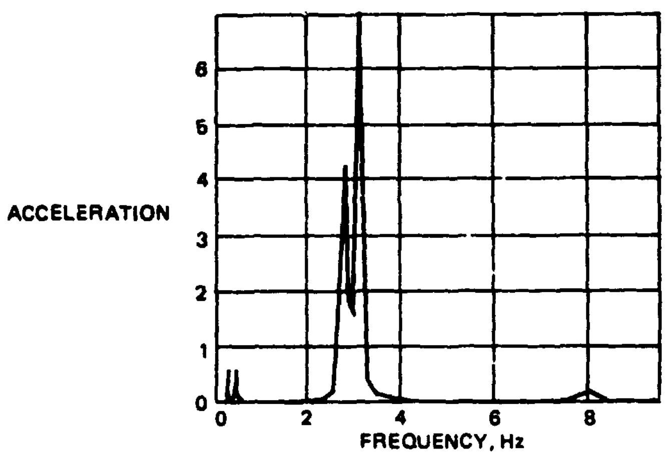
FIGURE 3.4-6 T-1A CONFIGURATION HARMONIC RESPONSE OF TDRS ANTENNA (SHEET 1 of 3)



D. X DIRECTION
2473-115D UNIT TORQUE ABOUT SPACECRAFT Y-AXIS

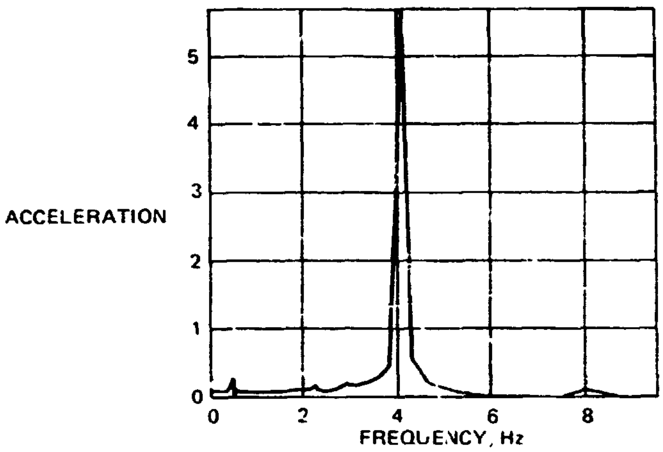


E. Y DIRECTION
2473-115E UNIT TORQUE ABOUT SPACECRAFT Y-AXIS



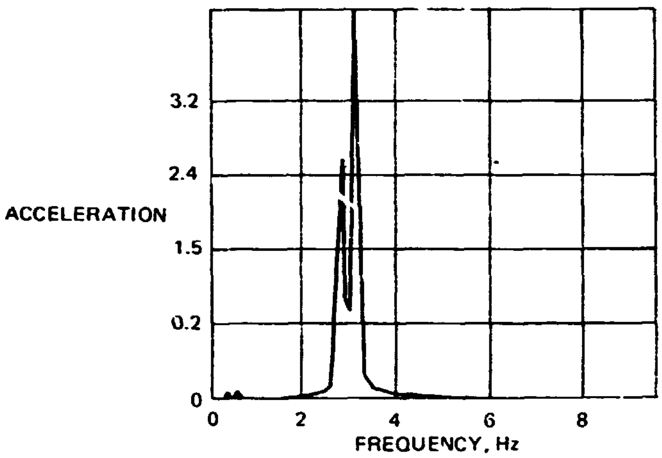
F. Z DIRECTION
2473-115F UNIT TORQUE ABOUT SPACECRAFT Y-AXIS

FIGURE 3.4-6 T-1A CONFIGURATION HARMONIC RESPONSE OF TDRS ANTENNA (SHEET 2 of 3)



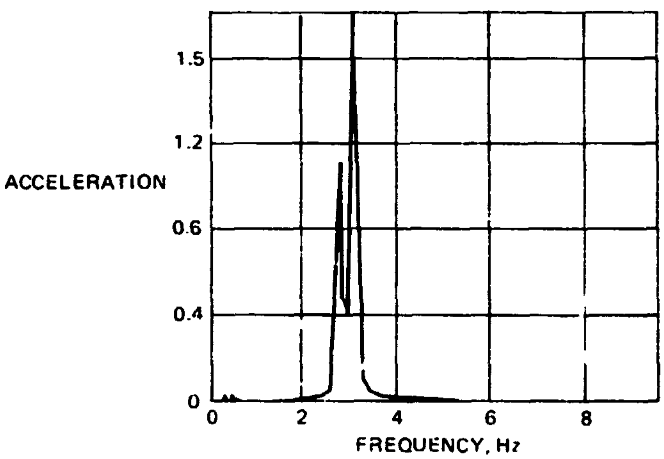
2473-115G

G. X DIRECTION
UNIT TORQUE ABOUT SPACECRAFT Z-AXIS



2473-115H

H. Y DIRECTION
UNIT TORQUE ABOUT SPACECRAFT Z-AXIS



2473-115I

I. Z DIRECTION
UNIT TORQUE ABOUT SPACECRAFT Z-AXIS

FIGURE 3.4-6 T-1A CONFIGURATION HARMONIC RESPONSE OF TDRS ANTENNA (SHEET 3 of 3)

to other than unit torques. Table 3.4-6 shows the maximum loads at the Solar Array and Antenna supports and the base of the extender tube for the unit input torque. These values could also be linearly scaled to other than unit torques.

3.4.3 References

1. NASA S-700-12, Mechanical System Specification for the Multimission Modular Spacecraft, GSFC, Rev. April 1976.
2. NASA-S-700-18, Multimission Modular Spacecraft (MMS) Subsystem Module Environmental Test Specification, GSFC, April 1976.

TABLE 3.4-6 SOLAR ARRAY AND TDRS ANTENNA RESPONSE LOADS DUE TO UNIT INPUT TORQUES (1 IN-LB) AT S/C C.G.

	Element No.*	Frequency, (Hz)	Max Shear, (Lb)	Max. Moment In.-Lb	Max Torque In.-Lb
Input about X-Axis	1	0.328	0.0612	36.2	NFGL
	5	0.328	0.459	14.1	0
	8	5.73	0.214	40.3	NEGL
Input about Y-Axis	1	0.455	0.0371	14.6	9.40
	5	0.455	0.0208	3.28	0
	8	4.09	0.188	22.1	7.94
Input about Z-Axis	1	0.455	0.0617	24.3	15.6
	5	0.455	0.0346	5.47	0
	8	0.09	0.0770	9.06	3.26

*Element No. 1 is the base of the Solar Array Crank Support.
Element No. 5 is the Solar Array Extender Tube.
Element No. 8 is the base of the Antenna Boom.

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3.5 THERMAL ANALYSIS

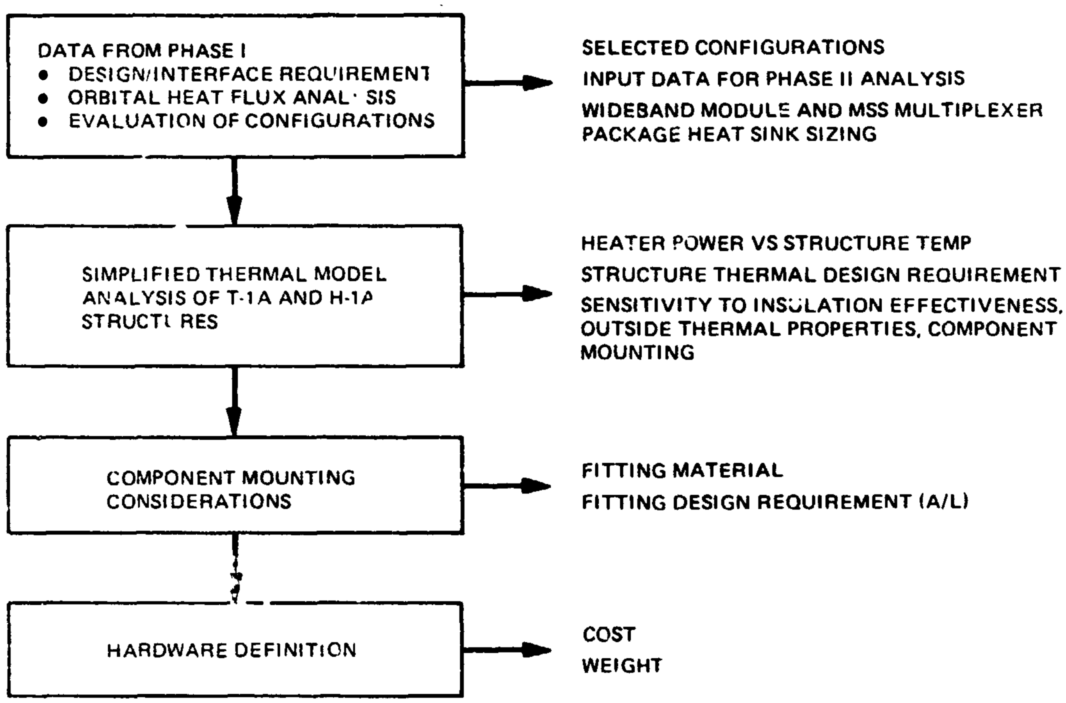
Thermal design constraints and interface requirements established (Figure 3.5-1) during Phase I were used to develop thermal evaluation criteria for the qualitative assessment of the eight initial design configurations These included:

- Thermally isolate components from module structure. Instruments and wideband module conductive coupling are to be less than 2 watts/°C
- Nominal operating temperature for instruments and wideband module to be 20°C
- MMS structure to be at same temperature as Instrument Module structure (no heat transfer across this interface)
- Maintain Instrument Module structure temperature between 0°C and 40°C (20°C nominal).
- Design orbit is sun synchronous; 705 Km altitude; 0930 DNTD.

Further evaluation using refined criteria indicated a preference for the T-1A and H-1A configurations based on thermal as well as other considerations leading to their selection for Phase II evaluation. Also during Phase I, orbital heat fluxes were determined. A generalized model with 30 surfaces was generated for application with all configurations. Analysis for the Wide Band Module and the MSS Multiplexer Electronic Package heat rejection/heat sink sizing was performed in support of the design effort.

An insulated design with heaters for structure temperature control was selected for the thermal design approach to be evaluated during Phase II. This approach provides simplicity combined with the reliability of positive control; its acceptability is determined by the magnitude of heater power required. During Phase I it was agreed that less than 100 watts was acceptable. Accordingly, the Phase II thermal analysis was directed at an evaluation of heater power as a function of temperature for the selected design configurations.

The use of a detailed thermal model was initially considered, but this approach was not consistent with the level of design detail available. In order to work in parallel with the design efforts, a more simplified thermal analysis was conducted to provide a timely evaluation of the design approach. This analysis proved sufficient for validation of the design approach. The thermal analysis approach is illustrated in Figure 3.5-1.



2473-97

FIGURE 3.5-1 PHASE II THERMAL ANALYSIS FLOW

3.5.1 Thermal Analysis of Structure and Assumptions

Simplified thermal analysis models were made for both the T-1A and H-1A structures. First, the surface areas for each structure were determined. Using the Orbital Heat Flux Analysis (Attachment 3.5-1), the total absorbed heat fluxes for the cold case were determined. Worst case assumptions were used to calculate the absorbed fluxes; the minimum fluxes were used for each surface independent of season; and, outside surface properties were selected to give the worst cold case (minimum α, maximum ϵ). In addition, the structure was considered to have an unobstructed view of space (no blockage). These assumptions were made to give the maximum heater power. Cold-case heater power was determined as a function of insulation effectiveness (ϵ_{eff} = 0.01 - 0.03) and outside thermophysical properties (α/ϵ = 0.1-1.0). The thermophysical properties used are summarized as follows:

Coating	α Range	ϵ Range	α/ϵ Cold Case	α/ϵ Hot Case
Silver Teflon	0.08-0.14	0.76	0.105	0.184
Aluminized Kapton	0.40-0.60	0.70-0.80	0.50	0.857
Black Paint	0.90	0.90	1.0	1.0

Heater power was also determined as a function of ΔT between mounted components and the structure, assuming five components each with a conductance of 2 watts/°C.

A hot-case analysis was made for the T-1A structure. This configuration was selected because it had the least heater power and would therefore have a higher tendency towards going to negative heater power (i.e., not requiring heater power to maintain temperature). Worst case assumptions were used, including maximum heat fluxes for each surface, independent of season, and insulation effectiveness equal to 0.01. Heater power was determined as a function of outside thermophysical properties.

In all cases the structure temperature range was 0°C to 40°C, with 20°C as the target nominal temperature. The nominal insulation effectiveness used was 0.02, this value being readily achievable.

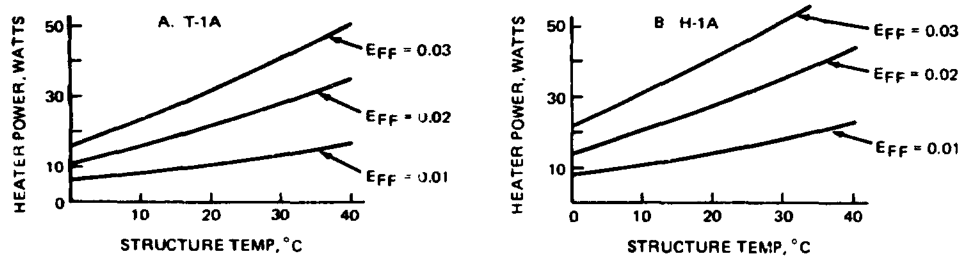
For all analyses, the interface between the Landsat structure and the MMS was considered adiabatic. This constraint was established by NASA during Phase I of the study.

3.5.2 Structure Analysis Results

The cold case analysis was conducted first to establish the heater power requirements. This analysis indicated that the outside properties of aluminized kapton were acceptable in terms of providing a positive margin of heater power of approximately 10 watts (See Figure 3.5.2a). Subsequent hot case analysis showed that this margin of control was reduced to 5 watts (Figure 3.5-3). This difference is due to the wide range in properties for aluminized kapton. The use of silvered teflon increases the minimum heater power control to 10 watts. Although the use of aluminized kapton may eventually prove to be sufficient, the use of silvered teflon has been assumed in the heater power numbers given in this document. The cold case analysis (Figure 3.5-4) shows that the use of silvered teflon adds approximately 10 watts of heater power. This is the penalty for providing more heater power control margin in the hot case.

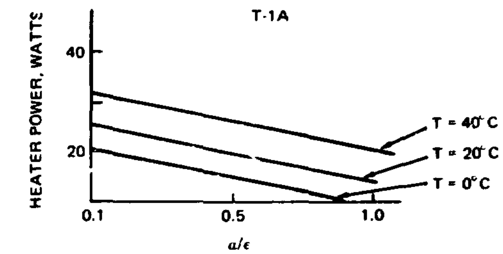
The cold case analysis shows that the nominal heater power required to maintain the T-1A structure temperature at 20°C is 31 watts (Figure 3.5-4). The term nominal means that the insulation effectiveness is 0.02 and the structure and mounted components are at the same temperature. The corresponding heater power for the H-1A structure is 39 watts (Figure 3.5-4b). This larger heater power requirement for the H-1A structure is due to the larger area (approximately 25% larger).

The effect of temperature difference between the structure and mounted components is shown in Figure 3.5-5 for the T-1A and H-1A configurations, respectively. This figure shows that heat leak to the mounted components has a significant impact on total heater power. The dashed curves in this figure shows the maximum heater power anticipated for heater sizing. For a 3°C temperature difference between the structure and components (conservative for the controls being considered), the maximum heater power for the T-1A and H-1A configurations is 72 and 83 watts, respectively. It should be noted that a reversal of this temperature difference results in a heat input to the structure, reducing the heater power control. For the hot case, as this temperature difference approaches 2°C, positive heater control may be lost. The present margin appears satisfactory, but if necessary, a controlled heat leak can be added to the design (increasing heater power) to maintain positive heater power control.



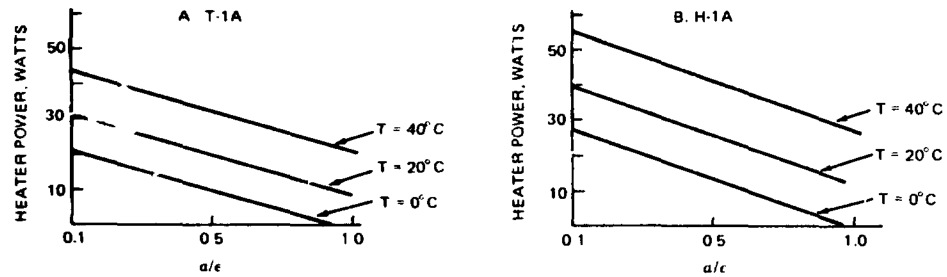
2473-98

FIGURE 3.5-2 HEATER POWER AS A FUNCTION OF INSULATION EFFECTIVENESS



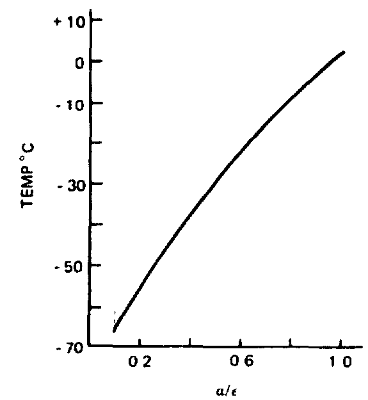
2473-99

FIGURE 3.5-3 HEATER POWER AS A FUNCTION OF SURFACE PROPERTY (HOT CASE)



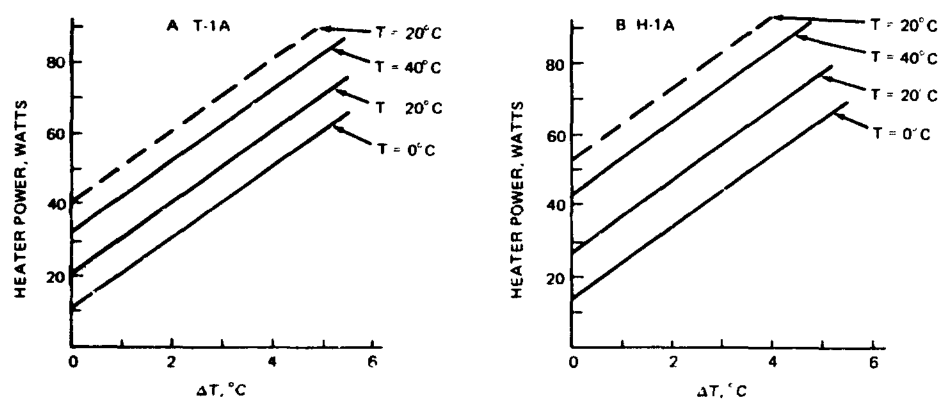
2473-100

FIGURE 3.5-4 HEATER POWER AS A FUNCTION OF SURFACE PROPERTY (COLD CASE)



2473-102

FIGURE 3.5-6 STRUCTURE TEMPERATURE W/O HEATER POWER



2473-101

FIGURE 3.5-5 HEATER POWER VS ΔT

The sensitivity to design parameters is also shown in the data. Figure 3.5-2 gives the variation of heater power with insulation effectiveness. For a variation of ±0.1 in insulation effectiveness the heater power changes ±10 watts for the T-1A configuration and ±13 watts for the H-1A configuration. Figure 3.5-4 shows heater power as a function of outside thermophysical properties (α/ϵ). The sensitivity to α/ϵ changes of ±0.1 is approximately ±3 watts.

Local transient orbital temperature variations is a subject for detailed analysis. In lieu of this, a simplified lumped parameter analysis was made to provide indicative temperature swings. For a 35 minute dark period, with zero external heat flux, the structure temperature change was less than 2°C. This temperature change is well within the heater control circuit capability, and would be dampened out.

Analysis of module structure temperature without heater power is shown in Figure 3.5-6. This analysis is indicative of a total power failure or survival mode. The minimum structure temperature is -66°C.

3.5.3 Component Mounting Analysis

Conservative guidelines were established for component mounting fittings. Titanium material was selected for its low thermal conductivity. The TM, MSS, and WBM each have three fittings. The three fittings for each module are [illegible], being dependent on the load conditions. For each fitting, on each component [illegible] minimum cross-sectional area has been selected such that the total conductance of each component will be less than 2 watts/ °C. The original guidelines assumed a [illegible] value of 10 btu/hr/ft °F for titanium conductivity. The titanium material s[illegible] has a much lower value (4 btu/hr/ft °F). The thermal path has additional resistances which must be added to the minimum cross-section resistance. These combined factors will yield a total conductance significantly less than 2 watts/°C for each of these components. A similar design approach is anticipated for the TDRS and Solar Array mounting.

Although not specifically required, similar Solar Array Configurations were reviewed to obtain an indication of peak temperature. This review indicates that the Landsat/FRUSA configuration peak temperature should be in the range of 65°C to 71°C.

3.5.4 Thermal Hardware Configuration

The Landsat module structure is fully insulated with multilayer insulation. The insulation configuration selected is quite similar to that being used for the MMS. The insulation blankets have 17 layers, with [illegible] s of 0.3 mil aluminized kapton separated by dacron mesh. The inside layer is 3 mil aluminized kapton and the outside layer is 5 mil silver teflon. The blanket layers are perforated for venting. The estimated weight for the insulation is approximately 14 lb.

- Multi-Layer Insulation

  17-layer blanket with dacron mesh separators
  Outer layer - 5 mil silver teflon
  Mid layers 0.33 mil alum kapton
  Inner layer 3 mil alum kapton

  Approx weight (including all hardware): 14. lb

- Heater Circuits

  Heaters - Thermofoil strip type
  Weight (based on 80% coverage): 4. lb

  Thermostats - Solid State - can achieve ±1°C control weight (incl sensors) for 10 circuits: 2.3 lb

  Total weight of htr. circuits: 6.3 lb + Wiring

  Total weight - thermal control: 25 lb

Thermal control is achieved by heater circuits installed on the structure. Each circuit consists of thermo-foil strip heaters, a solid state thermostat, a thermistor, and override relays. Approximately ten heater circuits will be required. The weight estimate for these circuits is approximately 6.3 lb (plus wiring).

The thermostats considered are made by Frequency Electronics and have been used on GSFC satellites. The manufacturer has indicated that these thermostats can provide control to within ±1°C with an accuracy of ±0.1°C.

The total thermal subsystem weight for the Landsat module structure is estimated to be 25 lb.

3.5.5 Conclusions

Thermal control of the module structure can be achieved with a simple but reliable design using a fully insulated structure controlled with heaters. The design uses heater power at all times to achieve positive control, with an accuracy of ±1°C. The maximum heater power requirements, based on worst case analysis, are not excessive. The thermal subsystem weight (25 lb) is one third of the original GSFC estimate (Reference 8 of the st ment of work).

The thermal analysis shows that the cold case heater power (no heat exchange with mounted components) required to maintain the structure at 20°C is 31 watts for the T-1A configuration and 39 watts for the H-1A configuration. The maximum heater power for a 3°C temperature difference between the structure and mounted components, is 72 watts and 83 watts for the T-1A and H-1A configurations, respectively. This magnitude of heater power is acceptable, especially since the heater power is based on conservative assumptions. The design approach using heater control is, therefore, valid he structure design using shear panels is acceptable. It is unnecessary to use a truss design to reduce the heat transfer area.

The significant results of the thermal analysis are:

- Shear member structure is acceptable
- Heater power nominal (struct at 20°C), T-1A: 31 watts, H-1A: 39 watts
- Outer surface SBT for positive heater control (Penalty: 10w)
- Maximum heater power (insul eff: 0.03, struct/instr. gradient: 3°C)
  T-1A: 72 watts, H-1A: 83 watts
- Minimum struct temperature w/o htr power: -66°C
- Thermal distortion: 14 $\widehat{sec}$/per °C
- Sensitivities: (1) Moderate - insul eff, & surf prop
  (2) High - comp mtg and gradient.

The heater power values are based upon using silver teflon for the outside layer of the insulation blankets. The coating was tentatively selected to provide a larger margin of heater power control in the hot case. Detail design analysis may show that

aluminized Kapton outside properties are acceptable, thereby reducing the heater power requirement by approximately 10 watts.

The thermal design configuration consists of a fully insulated Instrument Module structure using multilayer insulation and heater circuits controlled by solid-state thermostats. Thermal control to within 1°C is readily achievable. Heater power is used at all times to achieve positive control of temperature and gradients. This type of thermal control is similar to that used for gyros. The estimated thermal subsystem weight is less than 25 lb.

The thermal analysis conducted in this study is consistent with the level of concept development thus far achieved. The next step should include detailed thermal modelling including blockage effects and thermal models of the instrument packages. Towards the end of Phase II the build-up of such a model for the T-1A configuration was started. This model can be used to refine the analysis, locate heaters, and evaluate transient effects. Analysis of appendages should be added as their designs develop.

ATTACHMENT 3.5-1
ORBITAL HEAT FLUX ANALYSIS

1) Generated a surface flux model which represents the instrument module surfaces.

2) Using the Grumman orbital heat flux computer program, generated transient and orbital average heat fluxes (direct solar, earth albedo, and earth IR), for the specified orbit parameters.

3) Secondary effects such as blockage of albedo and earth IR and reflection of direct solar, albedo and earth IR are not initially included.

4) Thermal environment constants

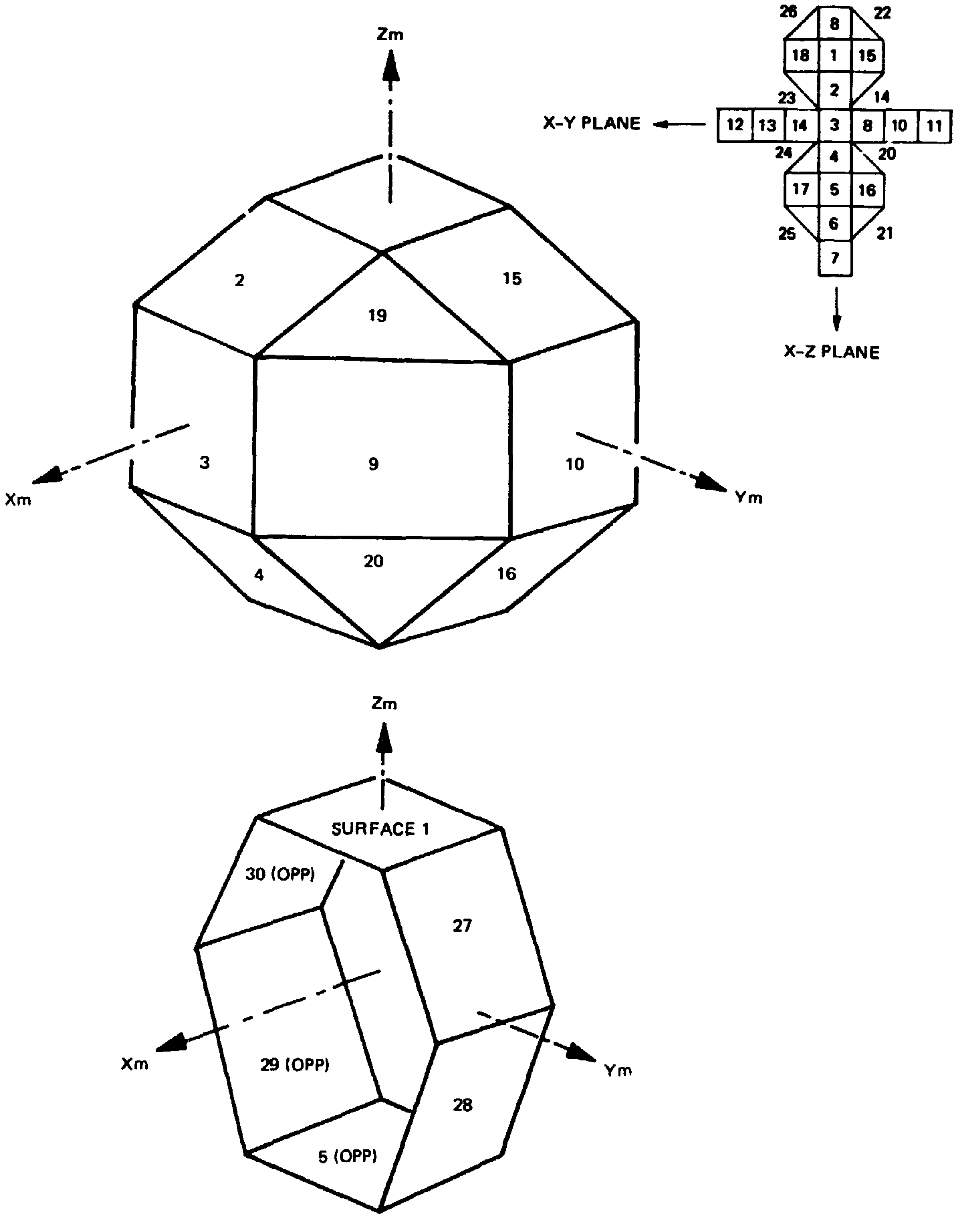
Solar Constant:	Vernal equinox	= 430 btu/hr/ft^2
	Winter Solstice	= 444 btu/hr/ft^2
	Summer Solstice	= 415 btu/hr/ft^2
Albedo Constant	0.30	
Earth Emission	75 btu/hr/ft^2	

5) Orbit Parameters

Altitude:	705 km (380.7 n mi) - Circular
Inclination:	98.20° - South - Heading
DNTD:	0930

Times of Year: Vernal Equinox, Winter Solstice, Summer Solstice

Attachment 3.5-1



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ORBITAL FLUX MODEL SURFACES

Attachment 3.5-1

9:30 AM, 705 KM, VERNAL EQUINOX

FINAL OUTPUT SURFACE	SOLAR	AVERAGE FLUX IN BTU/HR/SQ FT ALBEDO	EARTH	OS+A+E
1	17.665	26.404	80.527	104.596
2	42.803	19.282	44.192	106.277
3	83.766	7.681	17.611	109.058
4	107.713	0.610	1.399	109.722
5	109.513	0.0	0.0	109.513
6	107.699	0.610	1.399	109.708
7	83.709	7.675	17.611	108.995
8	42.736	19.274	44.191	106.202
9	8.604	6.908	17.607	33.118
10	0.0	6.585	17.611	24.196
11	8.516	6.904	17.606	33.025
12	132.978	8.429	17.606	159.014
13	175.976	8.742	17.611	202.329
14	132.971	8.432	17.606	159.010
15	0.0	17.903	44.191	62.094
16	8.941	0.450	1.398	10.790
17	189.352	0.759	1.398	191.509
18	68.420	20.543	44.193	133.254
19	9.616	18.310	44.193	72.119
20	38.874	0.499	1.400	40.773
21	39.032	0.499	1.400	40.931
22	9.392	15.305	44.192	71.889
23	62.113	20.248	44.193	126.553
24	162.346	0.717	1.400	164.463
25	162.512	0.717	1.400	164.629
26	61.897	20.243	44.192	126.332
27	0.0	13.876	35.126	49.002
28	0.0	1.577	4.862	6.539
29	198.320	2.541	4.862	205.723
30	108.471	16.747	35.126	158.345

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9:30 AM, 705 KM, WINTER SOLSTICE

FINAL OUTPUT		AVERAGE FLUX IN BTU/HR/SQ FT		
SURFACE	SOLAR	ALBEDO	EARTH	OS+A+E
1	18.513	27.107	60.527	106.147
2	44.289	19.794	44.192	108.275
3	86.290	7.885	17.012	111.787
4	110.650	0.626	1.399	112.675
5	112.427	0.0	0.0	112.427
6	110.631	0.626	1.399	112.655
7	86.211	7.880	17.611	111.703
8	44.197	19.788	44.191	108.176
9	8.195	7.078	17.806	32.879
10	0.0	6.743	17.611	24.354
11	8.109	7.075	17.605	32.789
12	138.297	8.666	17.605	164.569
13	184.100	8.993	17.611	210.704
14	138.323	8.669	17.606	164.597
15	0.0	18.350	44.191	62.548
16	8.421	0.460	1.399	10.280
17	196.556	0.782	1.398	198.737
18	72.182	21.215	44.191	137.589
19	9.720	18.721	44.193	72.694
20	39.125	0.510	1.400	41.035
21	39.246	0.510	1.400	41.156
22	9.519	18.776	44.193	72.488
23	64.938	20.802	44.193	129.933
24	168.006	0.738	1.400	170.145
25	168.128	0.738	1.400	170.266
26	64.738	20.798	44.193	129.728
27	0.0	14.222	35.127	49.349
28	0.0	1.715	4.863	6.678
29	206.387	2.617	4.863	213.867
30	112.473	17.217	35.127	164.817

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9:30 AM, 705 KM, SUMMER SOLSTICE

FINAL OUTPUT SURFACE	SOLAR	AVERAGE FLUX IN BTU/HR/SQ FT ALBEDO	EARTH	OS+A+E
1	15.235	27.672	60.527	103.434
2	41.902	20.213	44.192	106.307
3	85.229	8.056	17.602	110.887
4	112.187	0.639	1.399	114.225
5	114.646	0.0	0.0	114.646
6	112.191	1.399	114.230	114.230
7	85.237	8.043	17.602	110.881
8	41.908	20.195	44.192	106.294
9	18.906	7.435	17.608	43.948
10	0.0	7.165	17.608	24.773
11	18.750	7.424	17.608	43.782
12	115.484	8.638	17.608	141.733
13	136.601	8.880	17.608	163.089
14	115.322	8.646	17.608	141.576
15	0.0	19.106	44.192	63.298
16	21.907	0.508	1.399	23.814
17	166.861	0.754	1.399	169.014
18	48.200	21.286	44.192	113.677
19	12.630	19.436	44.193	76.259
20	53.234	0.549	1.400	55.182
21	53.282	0.549	1.400	55.230
22	12.766	19.423	44.192	76.381
23	51.294	20.978	44.193	116.465
24	151.347	0.723	1.400	153.469
25	151.218	0.722	1.400	153.340
26	51.254	20.984	44.192	116.411
27	0.0	14.900	35.126	50.025
28	0.057	1.861	4.862	6.780
29	168.002	2.549	4.862	175.413
30	68.647	17.183	35.125	120.956

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3.6 MASS PROPERTIES

A weight analysis was conducted for the two configurations, T-1A and H-1A, providing the centers of gravity and inertias of the designs for both the launch and deployed conditions.

The scope of the study limited the effort to a preliminary analysis but it is supported by structural, thermal and dynamic analyses and, with the 15% contingency growth allowed, provides realistic instrument module gross weights.

The data presented for each configuration includes a summary weight statement and mass properties for the instrument module launch and in-orbit conditions and for the observatory launch condition.

3.6.1 Weight Estimates

The weights of the structure and associated equipment were estimated based on the considerations discussed in the following paragraphs.

3.6.1.1 Structure

Two approaches were used to determine the structural weight:

- Utilize Grumman's RAVES Program 535 to establish an idealized weight for the primary structure
- Estimate the weight from the structural arrangement drawings LSD 127 and LSD 128.

The first approach is fully described in the structural analysis section of this report. It produced an idealized weight of 74.9 lb for the T-1A primary structure. This weight requires an additional non-optimum factor, and although Shanley suggests a factor > 2.0, Grumman's experience with other space and aircraft programs indicates the use of 2.5 is appropriate for this design. This gives a primary structure total of 187 lb.

The H-1A was not similarly analyzed since it was assumed to be comparable to the T-1A, the rationale being that the larger structure of the H-1A would be offset with a reduction in loads.

The second approach was to make a detailed weight estimate based on the sizes indicated on drawings LSD 127 and LSD 128. This preliminary layout stage produced a

weight of 170 lb for H-1A structure and 150 lb for T-1A. A layout factor of 1.25 was applied (a factor based, once again, on Grumman's experiences with previous programs). The results are, therefore, 212 lb for H-1A and 188 lb for T-1A.

The support structure which comprises the array, antenna and equipment supports could only be estimated from the structural arrangement drawings.

The equipment fittings are estimated in titanium for thermal considerations. Unlike many designs which require overstrength supports to provide adequate heat dissipation paths, this design requires a poor conduction path. Most of the equipment mountings are tied directly into primary structure with the result that the support/equipment ratio is estimated at 3.5% for T-1A and 4.3% for H-1A.

The overall unit weight for the structure is 3.5 lb/ft^2 of wetted area for the T-1A configuration and 3.2 lb/ft^2 for the H-1A configuration. This differential reflects the difference in module sizes and the lighter weight of the H-1A thematic mapper.

3.6.1.2 Solar Array

Table 3.6-1 gives details of the weight estimate with comments on the basis for the estimate. The stem device weight of 18 lb each is comprised of a 2 in. diameter tube for dynamic considerations, weighing 10 lb each and the deployment mechanism weight of 8 lb, using a root drum model.

3.6.1.3 Harness

No weight estimate has been made for the wiring and connectors, since no data is available to provide a basis for an estimate. The 75 lb allocation appears to be entirely adequate.

3.6.1.4 Thermal Control

The basis for the unit weight of insulation used in these estimates is detailed in the thermal control section of this report. At 0.2 PSF the difference in area between T-1A and H-1A (65 ft^2, 78 ft^2) gives a total weight differential of 3 lb. The active control, consisting of heaters, thermostats, relays, etc., is the same for both configurations and weighs 11.0 lb, based on details given in the thermal control section.

3.6.1.5 Communication

Table 3.6-2a gives the TDRSS antenna weight breakdown for the two configurations. The differences are principally due to the mast length variation. The weight estimate is for a 10 Hz system based on line 2 of Table 3.6-2b.

3.6.2 Summary Weight Statement

During the Phase I portion of the study, eight configurations were evaluated in the launch (stowed) condition. The results of this analysis are presented in Table 3.6-3. The effects of solar array position, antenna mast length and contingency weight are shown in Table 3.6-4 for the on-orbit (deployed) configuration of the selected configurations H-1A and T-1A. Table 3.6-5 presents the worst-case cg effects with the MMS included. (The cg of the 1773 lb MMS was estimated to be on the optical axis (Y=Z=0), 2 feet below station 565.2).

A comparison of the two selected configurations (Table 3.6-6) shows a weight difference of 400 lb in favor of the H-1A. Both configurations, however, are within the gross observatory weight of 3670 lb. The net difference of 348 lb between the H-1A and T-1A (-400 lb mapper and +52 lb subsystem) when adding a 15% contingency/growth factor results, coincidentally, in the 400 lb difference.

The weight is broken down into six sections. Mechanisms for the array and the TDRSS antenna are given with their respective components. The weight margin and assigned cg's and inertias reflect an instrument module margin rather than an observatory margin. The result is a substantial shift in longitudinal cg's and some inertia changes to the total observatory values.

3.6.3 Mass Properties Summary - Launch Condition

Mass properties including cross products for major components and for complete instrument modules are shown in Tables 3.6-7 and 3.6-8 for the T-1A and H-1A configurations, respectively. They have been combined with the MMS in Tables 3.6-9 and 3.6-10 for a 3670 lb observatory. The MMS is estimated to have a vertical cg of 34 in. below the user interface and radii of gyration of 34.7 in. about the Y-axis and 318. in. about the Y and Z-axis. The array weight has been considered in two parts for purposes of better calculating the mass properties, rather than in three parts as shown in the weight summary.

3.6.4 Mass Properties Summary - In Orbit

Tables 3.6-11 and 3.6-12 present the mass properties of the instrument module with the solar array and the TDRSS antenna fully deployed for the T-1A and H-1A configurations, respectively. The data are given for four different positions of the array as indicated in the diagram accompanying the Tables. The minor variation due to the movement of the antenna has been ignored by maintaining a central position throughout the orbit. The mass properties of the entire spacecraft at the maximum weight of 3670 lb and with the array in position A are also given.

TABLE 3.6-1 SOLAR ARRAY EQUIPMENT

Item	Weight, Lb	Comment
Solar Array	34	0.2 PSF Hughes FRUSA Program Data
Storage Roller	15	10 in. dia 80 in. x .10 in. Equiv Aluminum
Drive Motor Array	6	–
Spreader Bar	4	2 in. dia x 90 in. Aluminum
Stem Device (2)	36	Astro Research Data 0.2 in. OD Tube 0.010t
Stem Drive Torque Tube	3	1.5 in. x 87 in. x 0.063 x 1.25 Aluminum
Stem Drive Motor	5	–
Cushion Storage Roller	3	–
Orientation Mechanism		–
Strongback Tube	8	4 in. x 80 in. Aluminum
Swivel Locking Device	4	–
Swivel Shaft and Spring	1	–
Offset Crank	4	3 in. dia Aluminum
Orientation Drive Unit	25	Based on Drive Unit for ELMS Program
Control Electronic Unit	10	Hughes FRUSA Program Data
Instrumentation	2	
TOTAL	**160**	

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TABLE 3.6-2 COMMUNICATION EQUIPMENT
A. TDRSS ANTENNA

Item	Weight (Lb)	
	T-IA	H-IA
End Fitting	2	2
Inner Tube	10	25
Outer Tube	17	25
Upper Tube	10	14
Mast-Dish Attach Fitting	4	4
Extension Spring Shrink Cable	2	N/A
Gimbal Lock Mech	3	3
Rotation Cam	1	1
Drive Motor	5	5
Gear Reducer	2	2
Worm Gear Drive Mech		
Torque Tube	N/A	2
Lead Screw	N/A	2
TOTALS	56	85

B. ANTENNA MAST

	T-1A Configuration	H-1A Configuration
Baseline		
5 Hz System		
Tube Diameter	6 – 8 in.	6 – 10 in.
Tube "T"	0.063 in.	0.063 in.
Tube Weight	23 lb	39 lb
10 Hz System		
Tube Diameter	10 – 13 in.*	10 – 16 in.*
Tube "T"	0.063 in.	0.063 in.
Tube Weight	37 lb	64 lb

*If these diameters become mechanically unwieldy, other options which could be considered include material changes or dia/t variations, to obtain the required stiffness for a 10 Hz system (out of scope of this study).

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TABLE 3.6-3 PRELIMINARY CONFIGURATION ANALYSIS – SUMMARY

		CG (in)			Inertia lb(m)–in^2 x 10^{-6}			
Configuration	Wt, Lb	X	Y	Z	Ixx	Iyy	Izz	Ixz
H-1A	1497	42.71	-1.83	-4.66	.84	1.53	1.49	C.010
H-1A+cont	1897	40.03	-1.45	-3.68	1.02	1.89	1.84	-.009
T-1A	1897	50.27	-1.99	-2.76	1.06	2.15	2.37	0.044
H-1B	1497	40.05	-4.35	-1.97	0.79	1.43	1.53	-.052
H-1B+cont	1897	37.93	-3.43	-1.56	0.96	1.76	1.87	[illegible]
T-1B	1897	44.59	-2.45	1.39	1.01	2.34	2.55	.9
H-2	1497	0.75	43.85	3.56	1.37	0.67	1.74	022
H-2+cont	1897	0.59	40.93	2.81	1.73	1.04	2.10	0.023
T-2	1897	3.00	35.21	-2.08	1.32	1.14	1.74	-.047
H-3	1497	-1.43	-1.53	37.02	1.36	1.52	0.85	-.094
H-3+cont	1897	-1.13	-1.21	35.54	1.67	1.83	1.02	-.097
T-3	1897	0.81	-2.36	50.44	2.52	2.68	0.93	-.042

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TABLE 3.6-4 SELECTED CONFIGURATION SENSITIVITY SUMMARY

Configuration	Wt., Lb	CG (in) X	CG (in) Y	CG (in) Z	Inertia–lb(m)–in^{2}x10^{-6} Ixx	Iyy	Izz	Ixz	S.A. Position	Ant. Mast Length, Ft
H-1A	1497	37.53	−28.74	−23.25	9.37	4.20	6.96	0.100	A	10
H-1A+Δcont	1897	35.94	−22.68	−18.34	9.96	4.68	7.54	0.045	A	10
T-1A	1897	41.03	−18.84	−17.33	9.55	4.22	7.97	0.621	A	10
H-1A	1497	37.53	−28.74	− 2.80	10.16	5.00	6.96	0.543	B	10
H-1A+Δcont	1897	35.94	−22.68	− 2.21	10.59	5.31	7.54	0.537	B	10
T-1A	1897	41.03	−18.84	− 1.60	11.75	6.43	7.97	−0.036	B	10
H-1A	1497	42.74	−28.74	−13.02	8.36	3.78	7.56	0.423	C	10
H-1A+Δcont	1897	40.05	−22.68	−10.28	8.84	4.19	8.16	0.371	C	10
T-1A	1897	45.14	−18.84	− 9.66	9.26	3.96	8.00	0.571	C	10
H-1A	1497	27.31	−28.74	−13.02	8.36	4.15	7.93	0.122	D	10
H-1A+Δcont	1897	27.87	−22.68	−10.28	8.84	4.51	8.49	0.134	D	10
T-1A	1897	32.97	−18.84	− 9.66	9.26	[illegible].04	10.08	−0.230	D	10
H-1A	1497	37.53	−28.74	−33.43	14.32	9.16	6.96	0.398	A	20
H-1A+Δcont	1897	35.94	−22.68	−26.38	15.10	9.82	7.54	0.318	A	20
T-1A	1897	41.03	−18.84	−25.37	14.72	9.39	7.97	0.972	A	20

2473-88T

SOLAR ARRAY POSITIONS

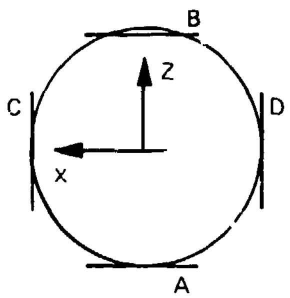

TABLE 3.6-5 WORST CASE ANALYSIS WITH MMS

Configuration	Wt, Lb	CG (in) X	CG (in) Y	CG (in) Z	Inertia lb(m)–in^{2}x10^{-6} Ixx	Iyy	Izz	Ixz	Antenna Mast Length, Ft
H-1A	3270	4.17	-13.16	-10.69	11.16	8.48	11.48	-1.061	10
H-1A+Δcont	3670	6.98	-11.72	-9.48	11.42	9.06	12.07	-.963	10
T-1A	3670	9.61	9.74	-8.96	10.83	9.14	12.94	-.412	10
H-1A	3270	4.17	-13.16	-15.30	16.58	13.91	11.48	-1.271	20
H-1A+Δcont	3670	6.98	-11.72	-13.63	16.89	14.53	12.07	-1.130	20
T-1A	3670	9.61	-9.74	-13.11	16.31	14.63	12.94	-.540	20

NOTE: CG's are about the instrument module axes.

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TABLE 3.6-6 SUMMARY WEIGHT COMPARISON

Code	Item	Weight, Lb					
		T-IA Configuration			H-IA Configuration		
1	STRUCTURE			230			250
	Primary		187			212	
	Supports		43			38	
2	Electrical			235			235
	Solar Array		160			160	
	Array	34			34		
	Mechanism	72			72		
	Orientation Mech and Cont	54			54		
	Harness		75			75	
3	Environment Control			24			27
	Insulation		13			16	
	Heater Circuits		11			11	
4	Instruments			798			398
	5-Band MSS		148			148	
	Thematic Mapper		650			250	
5	Communication			293			322
	Wideband Module		110			110	
	TDRS Antenna System		183			212	
	Dish	127			127		
	Mast and Mechanisms	56			85		
6	Contingency/Growth at 15%			237			185
	Instrument Module Total			1817			1417
	MMS S/C			1773			1773
	Margin			80			480
	Observatory Total			3670			3670

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TABLE 3.6-7 T-1A CONFIGURATION MASS PROPERTIES SUMMARY LAUNCH CONDITION

Item	Wt, Lb	CG (In.)			Inertia– lb(m)–in^{2}x 10^{-3}					
		X	Y	Z	I_{xx}	I_{yy}	I_{zz}	I_{xz}	I_{yz}	I_{xy}
1–Structure	230	17.9	-1.4	-3.6	85.8	102.5	112.9	-0.8	1.8	13.6
2–Electrical										
Array and Mech	131	45.0	-36.0	-6.0	2.2	78.1	78.1			
Orientation Mech	29	5.0	-30.0	-5.0	0.5	0.5	0.5			
Harness	75	25.0	0	0	20.0	32.5	32.5			
3–Environmental Control	24	23.0	0	0	15.0	11.6	11.6			
4–Instruments										
5-Band MSS	148	20.0	3.0	12.0	21.0	6.3	21.0			
Thematic Mapper	650	67.5	4.0	-6.3	457.7	307.1	307.1			
5–Communication										
Wideband Module	110	36.0	4.0	14.0	18.7	5.4	16.0			
TDRS Antenna										
Dish	127	116.0	-14.0	0	43.4	33.9	43.4			
Mast and Mech	56	75.0	-26.	-20.0	0.6	23.2	23.2			
6–Contig/Growth at 15%	237	52.1	-3.6	-2.3	35.6	35.6	35.6			
Instrument Module	1817	52.1	-3.6	-2.3	1043.7	2026.0	2232.4	-138.4	46.7	-45.2

CG's are about the instrument module axes.

2473-91T

TABLE 3.6-8 H-1A CONFIGURATION MASS PROPERTIES LAUNCH CONDITION

Item	Wt, Lb	CG (In.) X	Y	Z	Inertia lb(m)–in^{2}x10^{-3} I_{xx}	I_{yy}	I_{zz}	I_{xz}	I_{yz}	I_{xy}
1 Structure	250	24.9	–0.2	–7.1	94.5	120.9	145.5	–16.6	3.8	–2.0
2 Electrical	131	50.0	–8.0	–33.0	2.2	78.1	78.1			
Array and Mech	131	50.0	–8.0	–33.0	2.2	78.1	78.1			
Orientation Mech	29	5.0	–5.0	–25.0	0.5	0.5	0.5			
Harness	75	25.0	0	0	20.0	32.5	32.5			
3 Environ Control	27	25.0	0	0	11.3	9.2	9.2			
4 Instruments										
5-Band MSS	148	18.0	0	19.0	21.3	6.3	21.0			
Thematic Mapper	250	66.0	3.0	0	114.8	36.1	102.8			
6 Communication										
Wideband Module	110	40.0	0	19.0	15.6	6.0	16.9			
TDRS Antenna Dish	127	89.0	0	0	66.4	41.5	41.5			
Mast and Mech	85	50.0	–8.0	–32.0	1.4	51.9	51.9			
6 Contig/Growth at 15%	185	44.3	–0.9	–3.8	27.8	27.8	27.8			
Instrument Module	1417	44.3	–0.9	–3.8	727.0	1406.9	1203.4	–76.9	52.1	–11.0

NOTE: CG's are about the instrument module axes.

2473-94T

TABLE 3.6-9 T-1A OBSERVATORY LAUNCH CONDITION MASS PROPERTIES

Item	Wt, Lb	CG (In.) X	Y	Z	Inertias lb(m)–in^{2}x10^{-3} I_{xx}	I_{yy}	I_{zz}	I_{xz}	I_{yz}	I_{xy}
Instrument Module	1,817	52.1	-3.6	-2.3	1043.7	2026.0	2232.4	-138.4	46.7	45.2
Margin	80	52.1	0	0	12.0	12.0	12.0			
MMS	1,773	-34.0	0	0	2135.0	1791.5	1791.5			
Observatory	**3,670**	**10.5**	**-1.8**	**-1.1**	**3207.3**	**10623.3**	**10836.6**	**-313.0**	**54.3**	**-315.8**

CG's are about the instrument module axes.

2473-92T

TABLE 3.6-10 H-1A OBSERVATORY LAUNCH CONDITION MASS PROPERTIES

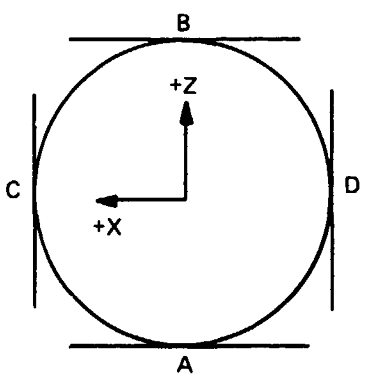
Item	Weight, Lb	CG (In.)			Inertia lb(m)–in^{2}x 10^{-3}					
		X	Y	Z	I_{xx}	I_{xy}	I_{zz}	I_{xz}	I_{yz}	I_{xy}
Instrument Module	1,417	44.3	−0.9	−3.8	727.0	1406.0	1203.4	−76.9	52.1	−11.0
Margin	480	44.3	0	0	72.0	72.0	72.0			
MMS	1,773	−34.0	0	0	2135.0	1791.5	1791.5			
Observatory	3,670	6.5	−0.4	−1.5	2947.0	8871.1	8656.1	−278.5	55.2	−39.4
NOTE: CG's are about the instrument module axes.										

2473-95T

TABLE 3.6-11 T-IA CONFIGURATION MASS PROPERTIES SUMMARY IN ORBIT CONDITION

Solar Array Position	Item	Weight, Lb	CG (In.)			Inertia lb(m)–in^{2}x 10^{-6}					
			X	Y	Z	I_{xx}	I_{yy}	I_{zz}	I_{xz}	I_{yz}	I_{xy}
A	Observatory	3670	4.9	−7.0	−8.4	9.45	11.90	12.05	−0.41	2.41	0.26
A	Instrument Module	1817	40.7	−14.1	−16.9	6.86	4.69	4.89	0.64	2.19	1.17
B	Instrument Module	1817	40.7	−14.1	−5.8	8.33	6.16	4.89	0.08	−0.56	1.17
C	Instrument Module	1817	46.3	−14.1	−11.4	6.62	4.87	5.31	0.73	0.81	−0.21
D	Instrument Module	1817	35.1	−14.1	−11.4	6.62	5.99	6.43	0	0.81	2.55
Note: CG's are about instrument module axes.											

Solar Array Positions



2473-93T

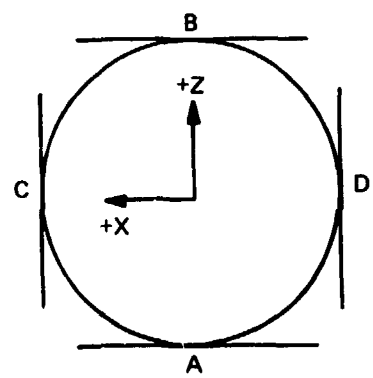
TABLE 3.6-12 H-1A CONFIGURATION MASS PROPERTIES SUMMARY IN ORBIT CONDITION

Solar Array Position	Item	Weight, Lb	CG (In.)			Inertia lb(m)–in^{2}x10^{-6}					
			X	Y	Z	I_{xx}	I_{yy}	I_{zz}	I_{xz}	I_{yz}	I_{xy}
A	Observatory	3670	1.4	−7.4	−9.4	10.89	12.14	10.55	−0.33	0.87	−0.15
A	Instrument Module	1417	31.9	−19.2	−24.3	7.85	5.39	3.99	0.66	0.47	0.68
B	Instrument Module	1417	31.9	−19.2	−10.0	9.56	7.10	3.99	0.28	−2.19	0.68
C	Instrument Module	1417	39.0	−19.2	−17.1	7.75	5.86	4.57	0.90	−0.86	−0.65
D	Instrument Module	1417	24.7	−19.2	−17.1	7.75	6.63	5.33	0.05	−0.86	2.00

Note: CG's are about the instrument module axes.

2473-96T

Solar Array Positions



3.7 INTERFACE ITEMS

The TM interface with the Instrument Module is primarily defined by the Thematic Mapper Interface Control Document, GSFC No. S-720-1. Design data, as well as other data that serve to update earlier concepts have been added as the result of a recently completed study conducted by Grumman.

It is the purpose of this preliminary listing to identify and document the significant study results that are of an interface nature; and, also to highlight the major interface requirements of the TM ICD that directly relate to the design intent of this investigation. The major headings of this preliminary listing include: Mechanical/ Structural Interface, Thermal Interface, and Environmental Interface.

3.7.1 Mechanical/Structural Interface

The mechanical/structural interface consists of items that are mostly defined by drawings; therefore, a list of preliminary drawings are included in this heading. Additional subheadings entitled Dimensional Limits, Mass Properties, TM Location and clear Fields-of-View, and TM Mounting and Alignment are provided with brief discussions to reference the drawing(s) that define the interface.

3.7.1.1 Interface Drawings

NASA/GSFC

Figure 10-a, GSFC No. S-720-1	Thematic Mapper Interface Dwg Concept No. 1
Figure 10-b, GSFC No. S-720-1	Thematic Mapper Interface Dwg Concept No. 2

TRW

AD78-33	Layout - Thematic Mapper - Revised Structural Concept

Hughes

PL 1162M102	Conceptual Interface - Thematic Mapper

GAC Dwg No.	Landsat Report Fig. No.	
LSD-120	2.7-1	T-1A Launch Configuration
LSD-119	2.7-2	H-1A Launch Configuration
LSD-124	3.1-2	T-1A On-Orbit Configuration
LSD-123	3.1-1	H-1A On-Orbit Configuration
LSD-127	3.1-4	T-1A Structural Arrangement
LSD-128	3.1-3	H-1A Structural Arrangement
MEM-101	3.7-3	Schematic Arrangement - Eqt Mtg Assy & Details

3.7.1.2 Dimensional Limits

The general configuration of the TM shall conform substantially to that shown in TRW Drawing AD78-33 or to Hughes Drawing PL1162M102; but in no case shall any of the instrument dimensions exceed the configuration envelope shown in NASA/GSFC Figure 10-a or Figure 10-b which appear in the ICD, GSFC No. S-720-1. Reproductions of these two figures are included as Figures 3.7-1 and 3.7-2.

3.7.1.3 Mass Properties

The TRW/TM instrument weight is 650 lb (295 kg), reference TRW Drawing AD78-33. The instrument center of gravity is also defined in this drawing.

The Hughes/TM instrument weight is 250 lb (114 kg), reference Hughes Drawing PL1162M102. The instrument center of gravity is also shown in this drawing.

3.7.1.4 TM Location and Clear Fields-of-View

The TRW TM installed location and orientation is shown in GAC Drawing LSD-120, LSD-124, and LSD-127.

The Hughes TM installed location and orientation is shown in GAC Drawing LSD-119, LSD-123, and LSD-128.

For either installation, the general arrangement and orientation of the TM onto the Instrument Module shall provide the TM with an unobstructed field-of-view for the radiative cooler on the cold side (+Y direction). The angular limits in the XY plane and ZY plane are defined in TRW Drawing AD78-33 or Hughes Drawing PL1162M102.

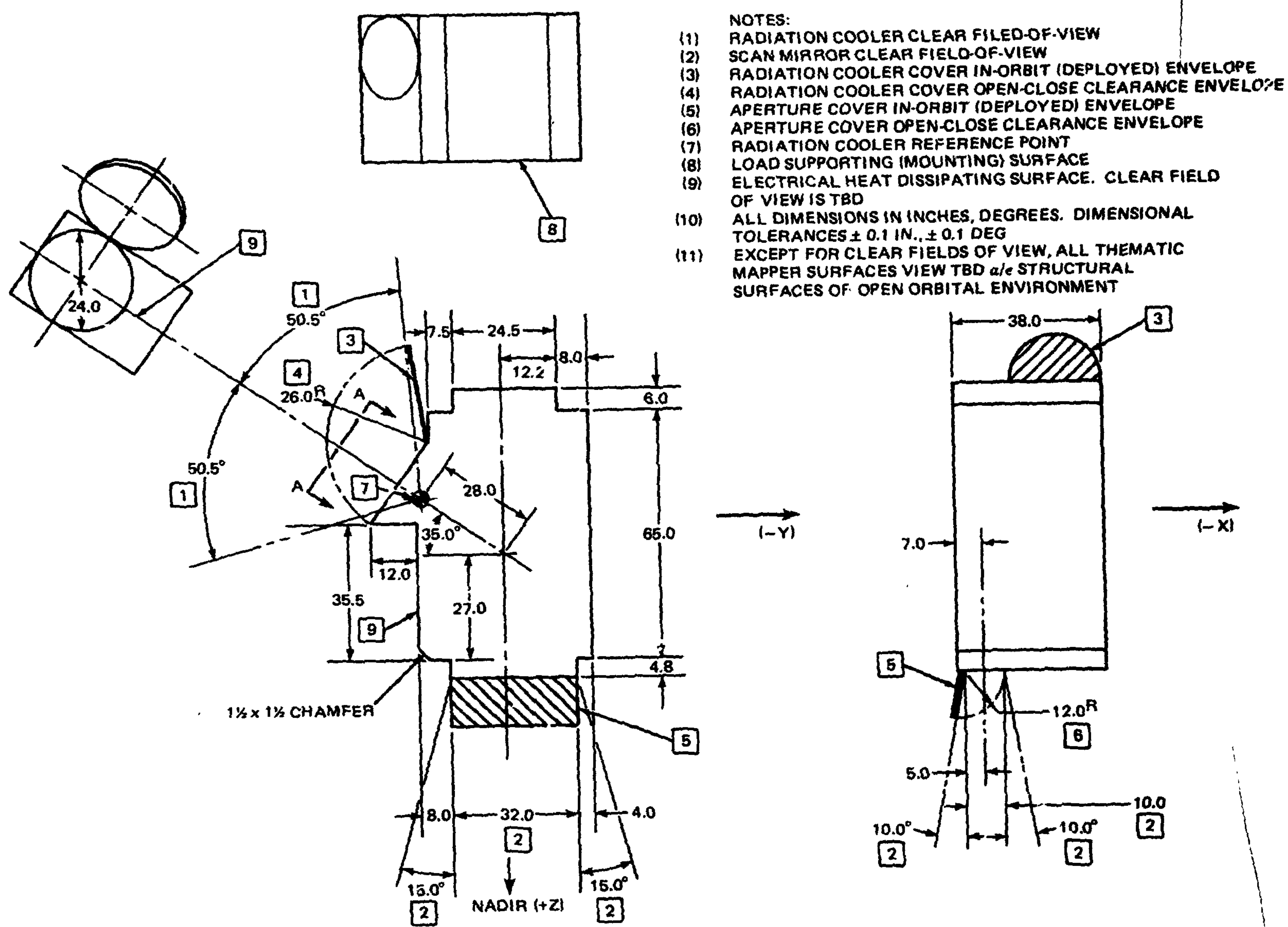
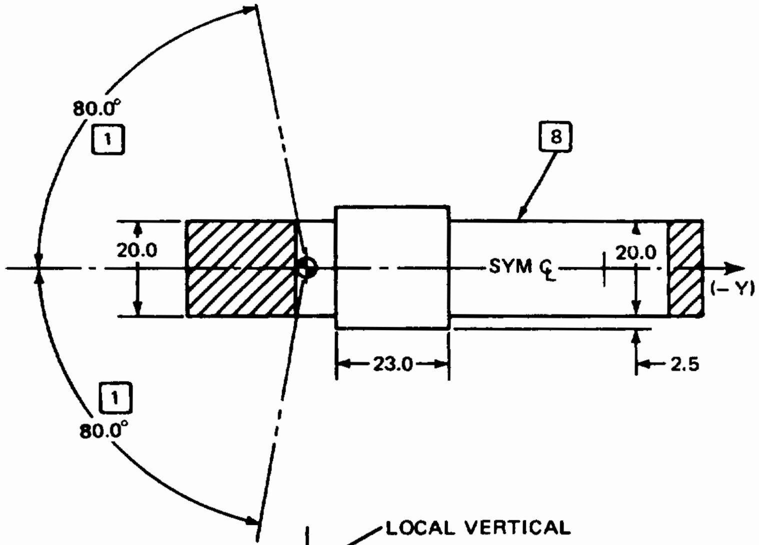


FIGURE 3.7-1 THEMATIC MAPPER INTERFACE DRAWING CONCEPT NO. 1

2473-76



NOTES

(1) RADIATION COOLER CLEAR FIELD-OF-VIEW

(2) SCAN MIRROR CLEAR FIELD-OF-VIEW

(3) RADIATION COOLER COVER IN-ORBIT (DEPLOYED) ENVELOPE

(4) RADIATION COOLER COVER OPEN-CLOSE CLEARANCE ENVELOPE

(5) APERTURE COVER IN-ORBIT (DEPLOYED) ENVELOPE

(6) APERTURE COVER OPEN-CLOSE CLEARANCE ENVELOPE

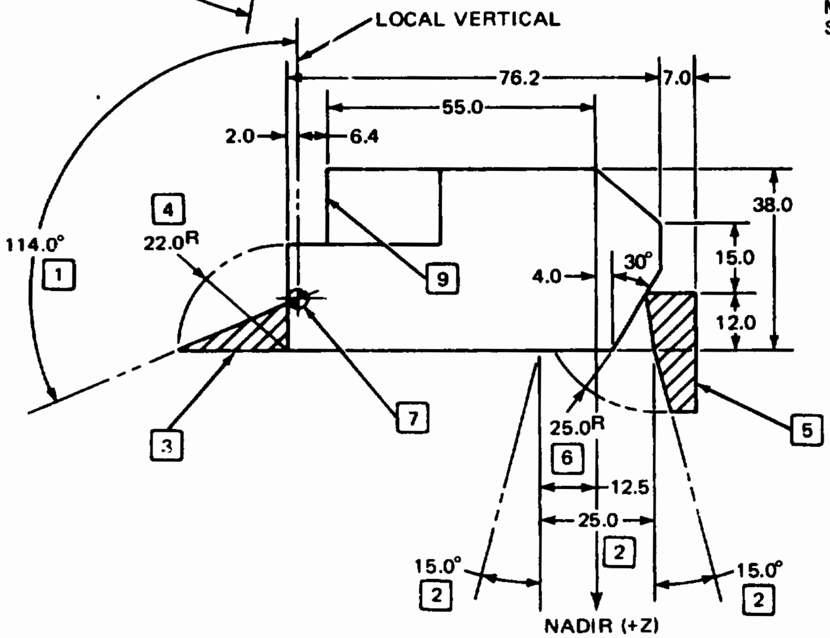
(7) RADIATION COOLER REFERENCE POINT

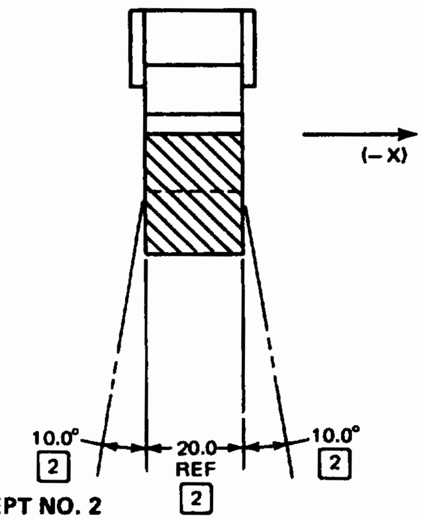
(8) LOAD SUPPORTING (MOUNTIN) SURFACE

(9) ELECTRICAL HEAT DISSIPATING SURFACE. CLEAR FIELD OF VIEW IS TBD

(10) ALL DIMENSIONS IN INCHES, DEGREES. DIMENSIONAL TOLERANCES ±0.1 IN., ±0.1 DEG

(11) EXCEPT FOR CLEAR FIELDS OF VIEW, ALL THEMATIC MAPPER SURFACES VIEW TBD α/ε STRUCTURAL SURFACES OR OPEN ORBITAL ENVIRONMENT





2473-77

FIGURE 3.7-2 THEMATIC MAPPER DRAWING CONCEPT NO. 2

Additionally, the earth viewing aperture of the TM scanner shall be provided with a clear field-of-view of ±15 degrees with respect to the nadir (+Z) in the cross-track direction; and ±10 degrees in the along-track direction.

3.7.1.5 TM Mounting and Alignment

The TM mounting shall be accomplished at three attachment points located on a surface of the Instrument Module, parallel to the YZ plane (Ref GAC Drawing LSD-127 and LSD-128). Ready access to the fasteners shall be provided in the design to facilitate installation, torquing, locking and removal of the instrument assembly. A design concept for the equipment mounting arrangement is provided in Figure 3.7-3.

At installation, alignment of the TM boresight referenced to the Instrument Module Z axis shall be within 0.2 degrees. The mounting installation shall be capable of sustaining launch and test loads without shifting more than 0.1 degrees from installation alignment.

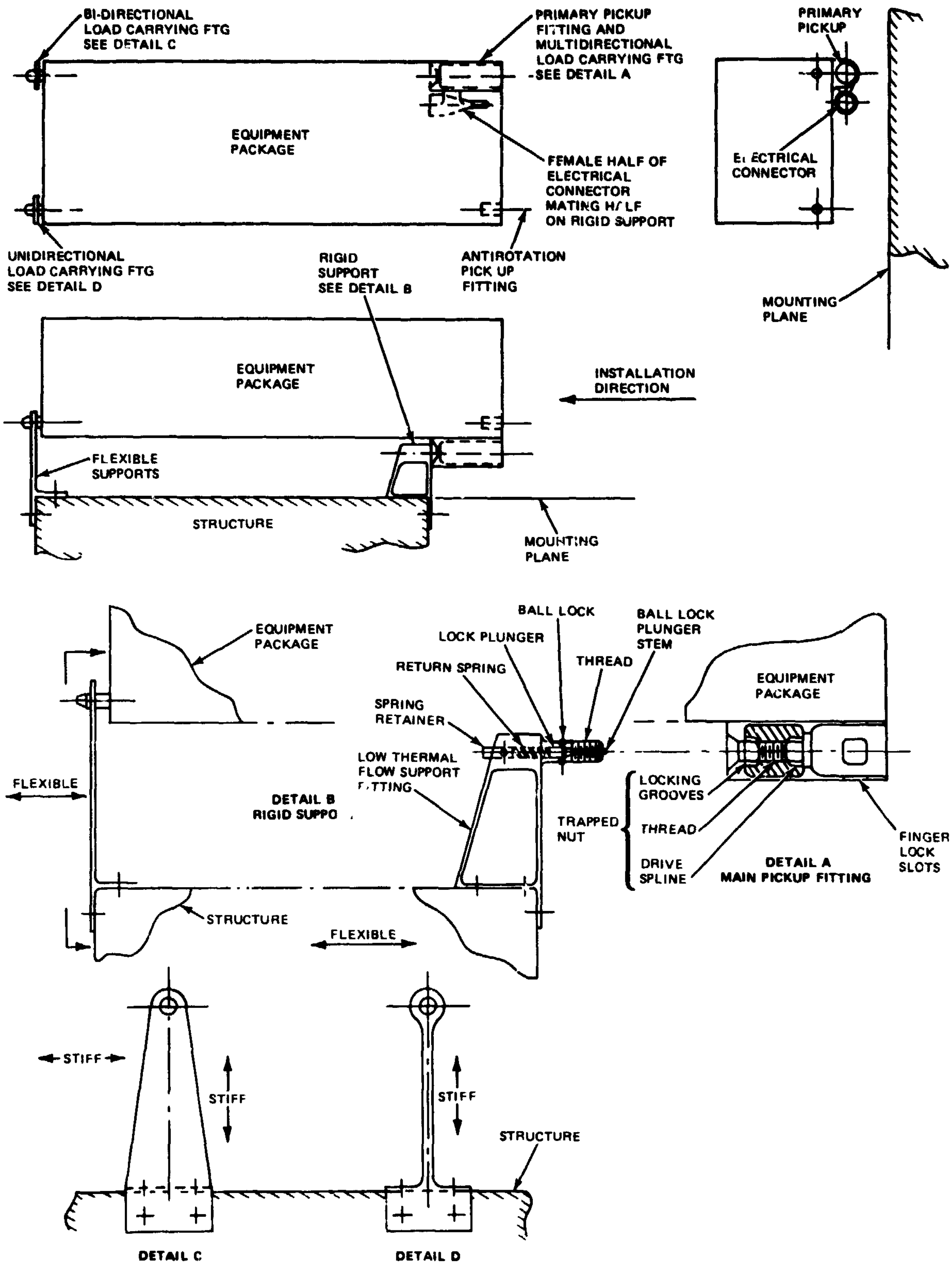
The mounting interface, instrument configuration, and operation shall easily allow the option of attaching the TM to a frame or mounting plate separate from the Instrument Module and capable of removal or installation in space via operations of the Module Exchange Mechanism (MEM).

3.7.2 Thermal Interface

The TM Instrument Module will be placed into a sun-synchronous, circular, near-polar orbit with a descending crossing of the equator scheduled at 0930 hours local time. The orbit inclination will be 98.2 degrees and the nominal altitude will be 705 km.

3.7.2.1 Thermal Environment

During on-orbit operation, the cold side of the Instrument Module will be facing the +Y direction with the +Z axis pointing at the spacecraft nadir. Direction of flight will be +X. The Instrument Module will be subjected to solar radiation, earth albedo, and also emitted energy from the earth. All surfaces of the Instrument Module structure will be provided with thermal insulation and coatings consisting of aluminized Kapton and silver-teflon in 17 layers for passive thermal control of the structure.



2473-78A
2473-78B

FIGURE 3.7-3 SCHEMATIC ARRANGEMENT EQUIPMENT MOUNTING ASSEMBLY AND DETAILS

In addition, active thermal control will be provided by the inclusion of thermostatically controlled heaters mounted on the structure to maintain the structure temperature at +20°C ± TBD.

Grumman will provide data defining the environmental heat fluxes incident on the TM external surfaces based upon the instrument configuration and outside thermophysical properties to be furnished by the TM contractor (TBD).

3.7.2.2 Heat Dissipation and Thermal Coupling

The TM shall control its internal thermal environment within the contractor and T.O. approved limits without dependence upon the Instrument Module as a thermal sink or source. The location and geometry of all TM passive cooler surfaces shall be specified. Conductive thermal coupling between the TM and the Instrument Module shall not exceed 2 watts/°C. The Instrument Module will be equipped with titanium fittings and hardware at the attachment points to minimize conductive thermal coupling to the TM.

3.7.2.3 Thermal Gradients

Thermal gradients within the TM and also within the Instrument Module structure shall be maintained within minimum practical limits consistent with TM performance requirements. A preliminary thermal analysis of the TRW/T-1A configuration indicates that the worst case thermally induced structural distortion will result in an alignment shift in the order of 14 arc sec/°C.

Acceptable limits of alignment shifts require further definition (TBD).

3.7.2.4 Thermal Nodal Model

A thermal nodal model of the TM and the Instrument Module shall be provided with sufficient detail to evaluate component temperatures, instrument package structural distortion, and thermal interchange between the TM and the Instrument Module structure. The model shall be fully documented, detailing all thermal properties, thermal couplings, heat inputs, and the assumptions and bases of the analysis (TBD).

3.7.3 Environmental Interface

The TM instrument and the Instrument Module to which it is mounted shall be designed to withstand the environmental conditions created by spacecraft launch, on-orbit operation, and Space Shuttle recovery and resupply operations. Additionally, in the Shuttle recovery mode, the environmental conditions of Shuttle re-entry and landing shall also be considered in the design.

Candidate launch vehicles include the Space Shuttle and the Delta 3910 vehicle.

3.7.3.1 Environmental Dynamics

The harsher environmental dynamics are primarily associated with the launch and boost phases of the mission with a lesser level occurring during the Space Shuttle operations. Residual, uncompensated momentums attributable to rotating machinery mechanisms, and the attitude control system will be in evidence during on-orbit operation but at significantly lower levels than that of the launch.

Environmental conditions that will be encountered include: vibration, shock, acceleration, and acoustic noise.

3.7.3.2 Protoflight and Flight Test Levels

In the protoflight test concept, structural dynamic stresses are imposed at 1.5 times the maximum expected flight stresses, while the time durations of the tests are limited to those expected in flight.

Protoflight and flight test levels for the TM are defined in Appendix A of the TM ICD, GSFC No. S-720-1. Tables and figures summarizing the test level for sinusoidal vibration, shock, acoustic noise, random vibration, and acceleration have been extracted from the ICD specification and reproduced as Tables 3.7-1 through 3.7-4 and Figure 3.7-4.

3.7.3.3 Dynamic Math Model

A dynamic math model of the Instrument Module for the T-1A configuration will be provided to analyze the dynamic loading produced by the launch and boost environment. This preliminary model will enable a gross prediction of fundamental frequencies and mode shapes for the Instrument Module, particularly at the TM mounting plane. Fundamental frequencies and generalized masses are shown in Table 3.7-5.

TABLE 3.7-1 SINUSOIDAL VIBRATION, PROTOFLIGHT AND FLIGHT TEST LEVELS

Axis	Frequency, Hz	Protoflight Level*	Flight Level	Sweep Rate, oct/min
Thrust (X-X)	5-15.5 15.5-50 50-200	0.5 in. (1.27 cm) ±6.0g ±2.0g	0.33 in. (.85 cm) ±4.0g ±1.3g	4.0
Lateral (Z-Z, Y-Y)	5-9 9-12 12-200	0.75 in. (2.00 cm) ±3.0g ±1.5g	0.5 in. (1.27 cm) ±2g ±1g	4.0

*Payload response limited to 1.5 x flight limit loads.

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TABLE 3.7-2 ACOUSTIC NOISE, PROTOFLIGHT AND FLIGHT TEST LEVELS

Octave Band Center Frequency, Hz	Protoflight Octave Band SPL (dB: re 20 $\mu N/m^2$)	Flight Octave Band SPL (dB: re 20 $\mu N/m^2$)
31.5	138	134
63	141	137
125	145	141
250	150	146
500	146	142
1000	139	135
2000	133	129
4000	130	126
8000	128	124
Overall	153	149

Duration: 1 Minute

Note: If it is not possible to attain the specified SPL's in the lower 3 octave bands (taking into account the allowable tolerances) due to test facility limitations, then a low frequency random vibration test will be performed in addition to the best attainable acoustic test for those low octave bands. The specification for this random vibration test is presented in Table 3.7-3.

2473-81T

TABLE 3.7-3 RANDOM VIBRATION, PROTOFLIGHT AND FLIGHT TEST LEVELS

Axis	Frequency Range, Hz	Protoflight PSD Level	Flight PSD Level	Duration
All	20-60 60-120 120-200	0.09 g^2 /Hz +6dB/oct 0.36 g^2 /Hz	0.04 g^2 /Hz +2.7dB/oct 0.16 g^2 /Hz	1 minute per axis
Overall Protoflight Acceleration Level = 6.7 g rms Overall Flight Acceleration Level = 4.5 g rms				
Note: Filter roll-off characteristics above 200 Hz should be 40 dB/oct or greater.				

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TABLE 3.7-4 PROTOFLIGHT ACCELERATION TEST

Acceleration Levels, g	
Thrust, x	Lateral
16.8 16.8	3.0 – Z Axis 3.0 – Y Axis
*Thrust and Lateral accelerations must be applied simultaneously.	
Shuttle Landing Loads	
Lateral – Z Axis Lateral – Y Axis	4.2 g 4.2 g

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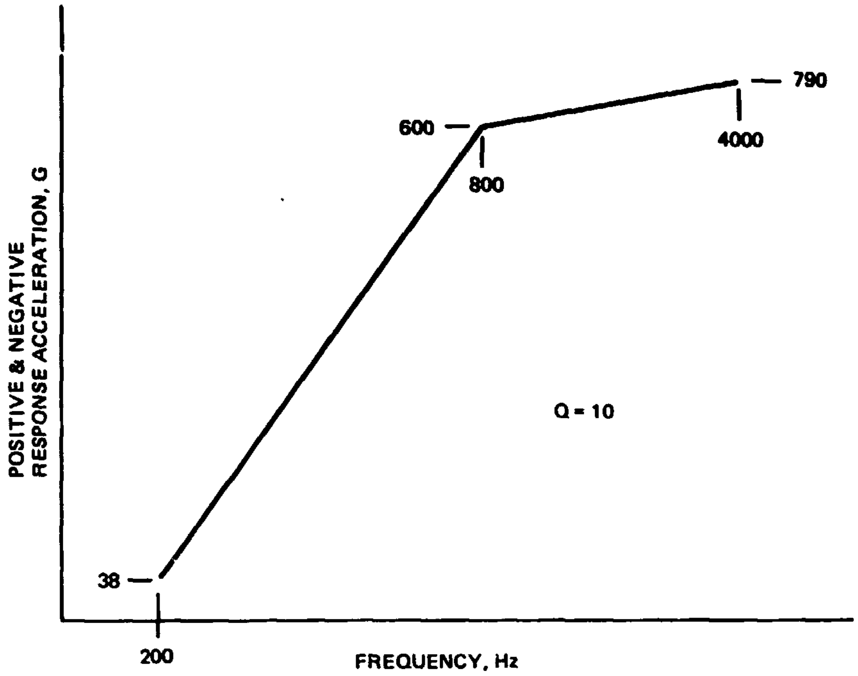

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FIGURE 3.7-4 PROTOFLIGHT SHOCK SPECTRUM

TABLE 3.7-5 T-1A INSTRUMENT MODULE LAUNCH AND BOOST CANTILEVERED MODES

Mode	Frequency, Hz	Gen Mass, Lb	Description
1	4.78	136	Antenna Dish Lat Translation (Y)
2	5.49	162	Antenna Dish Lat Translation (Z)
3	14.4	1071	Pedestal Lat Bending (Z)
4	22.5	799	Pedestal Lat Bending (Y)
5	29.9	322	Pedestal Torsion (0X)
6	32.5	198	WBM Vert Translation (X)
7	40.7	769	TM Vert Translation (X)
8	44.1	285	MSS Vert Translation (X)
9	52.3	267	Antenna Dish Vert Translation (X)
10	53.5	862	MSS and WBM Lat Translation (Y & Z)
11	60.0	688	SA Vert and Lat Translation (X & Y)

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SECTION 4

PROGRAMMATICS AND COSTS

The preparation of costing data for the Landsat Instrument Module was based on a number of elements which included: a Work Breakdown Structure (WBS) Dictionary; a Landsat Program Schedule including Development Test; various conceptual drawings and memos; and a Weight Statement for the Instrument Module estimated to the subsystem level.

4.1 COST ELEMENTS

The elements of cost are detailed below.

4.1.1 WBS Dictionary

The dictionary was prepared to describe each WBS element down to the subsystem level. It includes all equipments comprising the Instrument Module including sensors and related electronic equipment. Also, it identified other equipments which are assumed to be Government furnished.

The WBS dictionary is provided as Attachment 4-1 and a block diagram of these WBS elements is shown in Figure 4.1-1.

4.1.2 Program Schedule

A Landsat Program Schedule was prepared to show major development milestones, delivery, and launch dates for three flight articles. Included in this Program Schedule are time spans for the major elements of the development test program.

The Landsat Program Schedule is shown in Figure 4.1-2.

4.1.3 Other Items

Conceptual drawings for the Instrument Module and its various subsystems shown throughout this report were used extensively to determine configuration complexities, and also to serve as the data source for weight estimates. A Weight Statement estimated to the subsystem level was used as the basis for a Cost Estimating Relationship (CER).

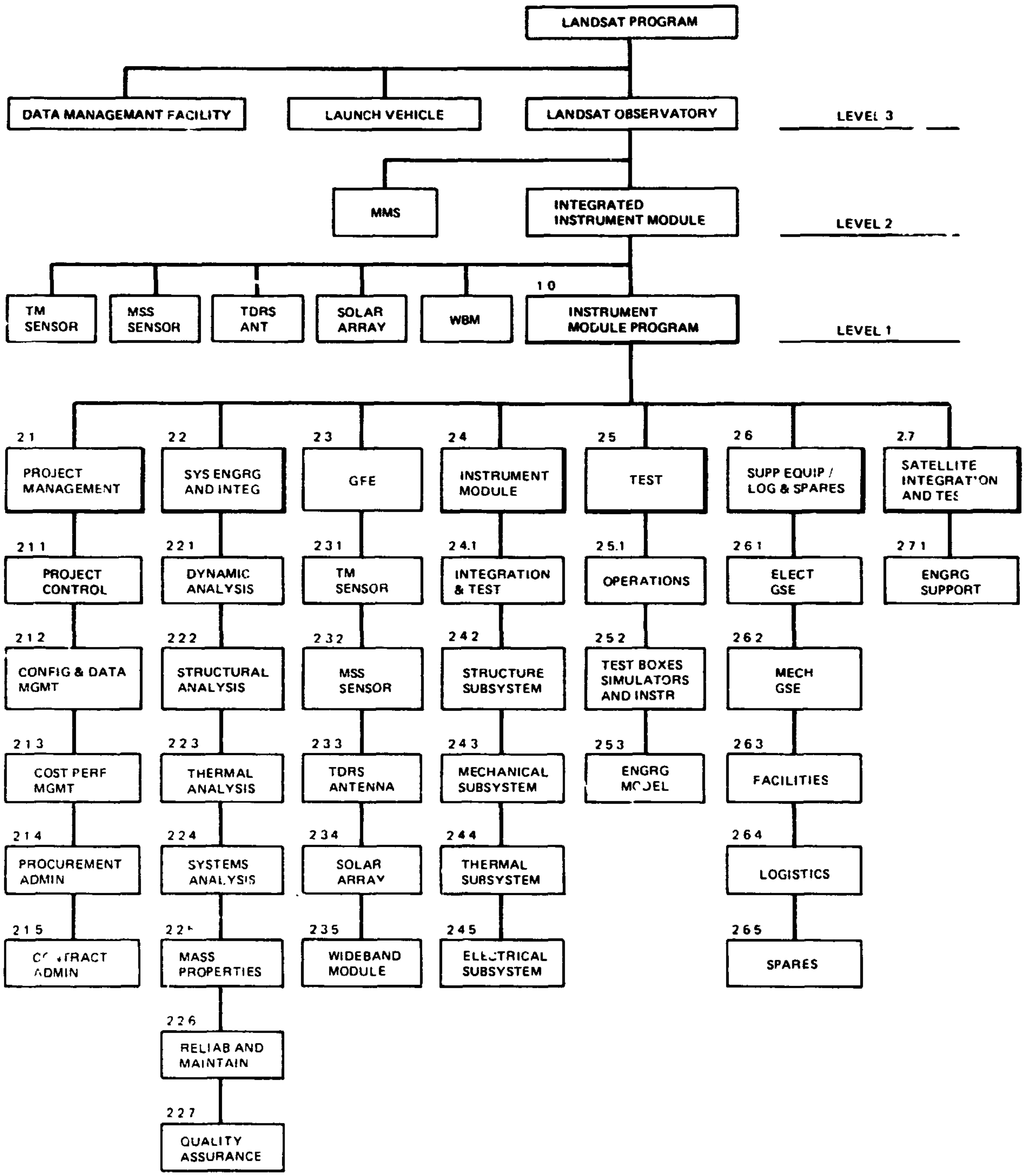

FIGURE 4.1-1 LANDSAT INSTRUMENT MODULE WORK BREAKDOWN STRUCTURE

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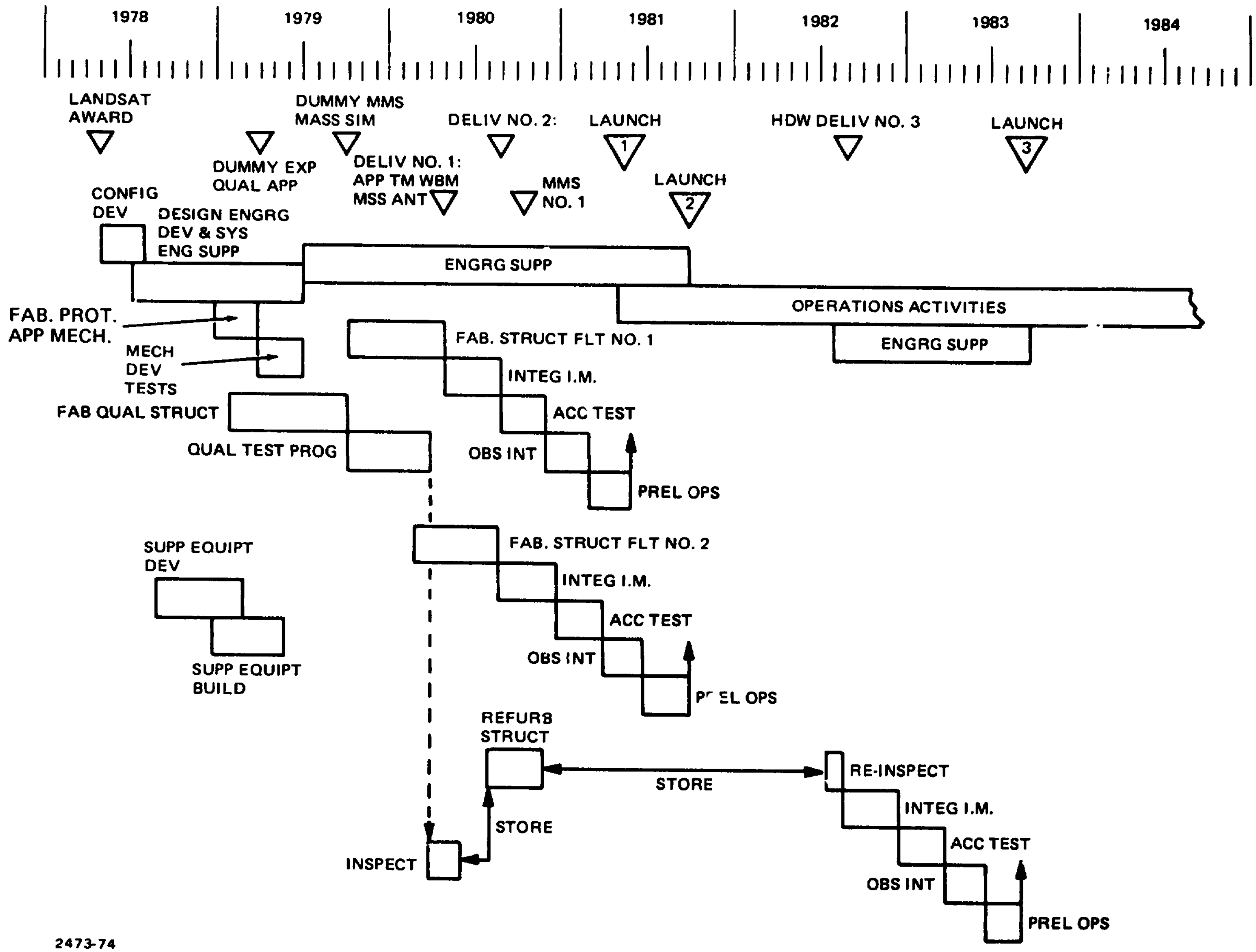

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FIGURE 4.1-2 LANDSAT PROGRAM SCHEDULE

4.2 GENERAL APPROACH

Initial cost estimates prepared early in July, 1976 covered all elements of the WBS including items identified as GFE. (These GFE items were excluded from subsequent estimates by direction of NASA/GSFC).

In the absence of a detailed configuration or any specific plan for development, test, and manufacture of the Instrument Module, a Cost Estimating Relationship (CER) based on weight was used in the preparation of all estimates. A cross check was made against initial estimates using a high level cost model. For the final estimate, a comparison was made by preparing manpower estimates for selected items of the Instrument Module design.

4.3 GROUND RULES AND ASSUMPTIONS

The following ground rules and assumptions were used in the preparation of the cost estimates:

- 1976 constant dollars, contractor fee excluded
- GFE assumed per WBS Dictionary was excluded from final estimates
- Level 1 Testing per WBS Dictionary
- Engineering Model included
- No multiple buy of flight modules
- Authority to proceed, April 1978
- First flight article available for Integration and Test, April 1980
- "Low" estimate assumes:
  - Availability of off-the-shelf equipment
  - Minimum requirements for data, controls, and reporting.

4.4 COST METHODOLOGY

The cost methodology is described below.

4.4.1 Source of Data

Actual costs for the OAO Program were used as the basis for an estimate that typifies a new development. Cost estimates derived from the EOS Study were used as the basis for an alternate approach which would have utilized a large proportion of off-the-shelf items.

4.4.2 Cost Estimating Relationships (CERs)

CERs were developed from each data source for application to subsystem weights, and other CERs were developed for test, system engineering, program management, and other areas of project work.

4.4.3 Cross Check of Estimates

In the first iteration of cost, a check was made by running a high level cost model for unmanned satellites. Although this model does not produce estimates at the WBS level of detail, model outputs produced a check on a total cost basis. By making a similar normalization of costs generated from the OAO derived CERs, a useful comparison was obtained. These costs include GFE, but not sensors. (Sensor estimates are beyond the scope of the cost model used.)

The second iteration included a comparison of the CER estimates with separately prepared estimates based on Manufacturing hours and Engineering manpower for Structure, Mechanical, and Test Elements of the WBS. This check provided an added level of confidence in the validity of the "High" and "Low" costs presented in Table 4.4-1.

4.5 CONCLUSIONS

As a result of this costing effort, it is believed that the total costs presented in Table 4.4-1 indicate the probable range of actual costs which would result from performing the Instrument Module Program in accordance with the definitions, ground rules, and assumptions that were utilized.

TABLE 4.4-1 LANDSAT ESTIMATED RANGE OF COST ($ MILLIONS)

		High		Low	
		Development	Prod/Unit	Development	Prod/Unit
1.0	,strument Module Program	(18.8)	(7.2)	(11.8)	(3.3)
2.1	Project Management	1.3	0.8	0.9	0.4
2.2	System Engineering/Integration	1.9	1.3	1.3	0.5
2.4	Instrument Module	(6.3)	(4.1)	(4.4)	(1.8)
2.4.1	Integration and Test	0.3	0.7	0.2	0.4
2.4.2	Structure Subsystem	1.8	1.0	1.4	0.4
2.4.3	Mechanical Subsystem	2.6	1.5	2.0	0.6
2.4.4	Thermal Subsystem	0.6	0.3	0.4	0.1
2.4.5	Electrical Subsystem	1.0	0.6	0.4	0.3
2.5	Test - Level 1	(6.9)	–	(3.6)	–
2.5.1	Operations	1.9	–	1.3	–
2.5.2	Test Boxes, Instrumentation	0.8	–	0.5	–
2.5.3	Engineering Model	4.2	–	1.8	–
2.6	GSE/Logistics/Spares	1.6	0.4	1.1	0.2
2.7	Satellite Integration/Test (Engrg/Mfg/Test/Support of Level 1)	0.8	0.6	0.5	0.4

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ATTACHMENT 4-1

LANDSAT INSTRUMENT MODULE
WORK BREAKDOWN STRUCTURE DICTIONARY

1.0 INSTRUMENT MODULE PROGRAM

All program effort to design, develop, test, fabricate, and deliver the Instrument Module, including support equipment, facilities and services, and support of Module integration with other elements of the Landsat Program.

2.1 PROJECT MANAGEMENT

All management effort required to plan, organize, direct, coordinate control and approve actions necessary to the accomplishment of program objectives. Included are the functions of the program manager, project control, configuration and data management, cost performance management, material procurement, and contract administration.

2.2 SYSTEM ENGINEERING AND INTEGRATION

All system engineering effort to analyze, define, integrate and control Instrument Module design and hardware, including the functions of dynamic analysis, structural and thermal analysis, mass properties, reliability, maintainability, quality assurance and safety. Excluded are: subsystem, component, and support equipment design, and specific support of design groups by analytical/specialist groups or disciplines.

2.3 SENSORS AND EQUIPMENT SUBSYSTEMS

All effort to procure and install sensors and equipment subsystems in the Instrument Module for the Landsat Mission. (These costs are to be identified as GFE).

2.3.1 Thematic Mapper (TM)

All TM sensor procurement and installation effort (GFE).

2.3.2 Multi-Spectral Scanner (MSS)

All MSS sensor procurement and installation effort (GFE).

2.3.3 TDRS Antenna

All effort to design, build, and acceptance test the TDRS Antenna subsystem and related hardware required for the Instrument Module. (Procurement costs for the TDRS Antenna dish, electronics, and drives are identified as GFE).

2.3.4 Solar Array

All effort to design, build, and acceptance test the Solar Array subsystem and related hardware required for the Instrument Module. (Procurement costs for the Solar Array assembly is identified as GFE).

2.3.5 Wide Band Module (WBM)

All effort to design, build and acceptance test the WBM subsystem and related hardware required for the Instrument Module. (Procurement costs for the WBM subsystem are identified as GFE).

2.4 INSTRUMENT MODULE

The effort to produce structure, mechanical and subsystem equipment hardware that contains and/or directly supports operating of the sensors and transmission of sensor data, and the integration and test of all hardware elements. Excluded are: engineering model, development test, and qualification test.

2.4.1 Integration and Test

All effort to assemble, integrate, checkout, and test the Instrument Module subsystem hardware ready for Customer Acceptance.

2.4.2 Structure Subsystem

All effort to design, build, acceptance test and install primary and secondary structural components which house and/or support sensors and related Instrument Module equipments for acquisition and transmission of sensor data.

2.4.3 Mechanical Subsystem

All effort to design, build, acceptance test, and install mechanical devices (such as motors, actuators, linkages, latches, etc.) which are required for the Instrument Module.

2.4.4 Thermal Subsystem

All effort to design, build and acceptance test the Instrument Module hardware that controls the thermal environment.

2.4.5 Electrical Subsystem

All effort to design, build, acceptance test, and install electrical power elements of the Instrument Module, including power distribution to Sensors, Wide Band Module TDRS Antenna and Solar Array.

2.5 TEST

All effort for development testing and qualification testing of the Instrument Module. Three levels of testing are defined:

Level 1 - Instrument Module including Thermal, Structural, Mechanical, and Electrical Subsystem elements.

Level 2 - Level 1 Subsystems plus Sensors, Wide Band Module, TDRS Antenna and Solar Array.

Level 3 - Level 2 Subsystems plus the Landsat modules that provide electrical power, stability and control and communication/data handling functions.

2.5.1 Test - Operations

All effort to plan, design, and build test fixtures and miscellaneous equipment, conduct and report tests at Instrument Module level in order to verify design concepts. Excluded are: subsystem level and acceptance test activities, test boxes and instrumentation, and the engineering model.

2.5.2 Test - Test Boxes and Instrumentation

All effort to design, build and test electrical/electronic boxes and related instrumentation for simulating inputs from the Landsat power module.

2.5.3 Test - Engineering Model

All effort to build and deliver to the test facility the Engineering Model of the Instrument Module which serves as the test article for performance of development and qualification testing. Basic design effort is excluded; maintenance engineering and engineering support of manufacturing are included in this WBS item.

2.6 GROUND SUPPORT EQUIPMENT, LOGISTICS, AND SPARES

The aggregate effort to design, build, and test GSE hardware and software required for handling, checkout, servicing and maintenance of the Instrument Module and related booms, antennas, etc., while not directly engaged in the performance of a mission.

2.6.1 GSE - Electrical

All effort to design, develop, build and test electrical equipment and software necessary to monitor and control the Instrument Module and its subsystems for functional, environmental and integrated testing for factory to launch operations.

2.6.2 GSE - Mechanical

All effort to design, develop, build and test mechanical equipment necessary to monitor and control the Instrument Module and its subsystems for functional, environmental and integrated testing for factory to launch operations. Also included are hardware items necessary for protection handling and transport of the Instrument Module and its subsystems.

2.6.3 Facilities

All effort required to design, develop, build and maintain special facilities required in support of the Instrument Module program.

2.6.4 Logistics

All effort required to provide publications and training data and equipment in support of the Instrument Module.

2.6.5 Spares

All effort required to analyze, plan, and implement the production and delivery of spare parts and components in support of the Instrument Module.

2.7 SATELLITE INTEGRATION AND TEST

All effort required to support the Instrument Module after delivery.

2.7.1 Engineering Support

All engineering effort to support satellite integration and test after Instrument Module delivery.

SECTION 5

CONCLUSIONS AND RECOMMENDATIONS

5.1 CONCLUSIONS

As a result of this study, two Landsat Instrument Module Configurations have been developed which satisfy all the orbital requirements for a sun-synchronous polar orbit earth observing satellite. The configurations are suitable for a Delta launch and can be adapted to shuttle operations for a retrieval mode.

In examining the two configurations developed, it becomes apparent that there are marked similarities between the Hughes and TRW versions, even though the experiments are substantially different. Although it is outside the scope of this study, there appears to be a potential to develop a common spacecraft design which would be compatible to both experiments. This point illustrates one of the major strengths of the configurations developed. The simple box-like aluminum structure, which forms the core of the chosen designs, has the capability of adapting to many different shaped and sized equipments with excellent potential for in-orbit access for refurbishment. In addition, the thermal design which isolates the structure from the equipment is quite amenable to equipment changes without substantial changes to the configuration. In conclusion, the designs though developed for a very specific set of ground rules, have a large degree of built-in flexibility to adapt to other requirements.

5.2 RECOMMENDATION

The study described in this report has met its goal of establishing and verifying two Landsat configurations. However, it must be remembered that the scope of the task was limited and that there are many areas that need to be explored before a full Landsat Program can be properly defined. Certainly in the detailed analysis and design substantiation areas many tasks must be performed. Probably more important initially, studies in the systems and subsystems areas should be carried on, much in the same vein as the study completed.

As potential candidates for follow-on studies, Table 5.2-1 lists many of these possibilities. In addition, one major area that must be addressed, is the development of proficient means of handling, managing, reducing and using the voluminous amount of data that Landsat will yield.

TABLE 5.2-1 POTENTIAL FOLLOW-ON STUDIES

STRUCTURAL DESIGN

- Design Changes to Accomodate Mods In Eqpt
- In-Depth Detailed Design
- Alternate Exp Fittings
- Common Structure Design
- Alternate Config Studies – Remove Trans Adapter Alt Matls, Titan, no repl in orbit, resupply IM, etc.
- General Structure Interfacing

MECHANICAL SYSTEM

- Detailed Sizing of Array
- Detailed Sizing Exp Mtg Fittings
- Develop Rigid Bar Ant Deploy System
- Develop MEM Adapter Tool Mechanism
- Develop Half. Stowed Shuttle Retrieval
- Examine Electrical Connector Interface
- Develop Detailed Back-up Eject Systems
- Gimballed Dir R.O. Antenna
- In-Orbit Refurb of Antenna/Array

STRUCTURAL ANALYSIS

- Dev-Detailed Model to Suit Selected Designs
- NASTRAN Above
- Thermal Distortion Analysis
- Integrate Anal. with MMS
- STRESS Anal. to Suit Final Designs

DYNAMIC ANALYSIS

- NASTRAN and Joint Effects
- Update Models and Calculate Forced Responses
- Dev Specific Exp Environments
- Perform Deploy Dynamics
- Interface Coupling with Booster/MMS for Loads

THERMODYNAMICS

- Dev Comprehensive Thermal Model and Conf
- Orbit Flux Anal. with Blockages
- Solar Array Model

MASS PROPERTIES

- Detailed Weight Analyses, CG/MOI

SUB-SYSTEM STUDIES

- Communications
- Data Handling
- Attitude Control
- Power Reqmts Study
- RF Reqmts
- Elect. Dist System and Addl Control Boxes
- Telem and Comm To Supt Deployments

SYSTEM STUDIES

- Mission Timeline for Baseline Orbit
- Sensor Operation Duty Cycle
- Observatory Power Duty Cycles
- Orbit Operation and Geom Des Effects
- Sys Study – Level of Refurbishment
- Sys Study – Shuttle of Refurb Operation
- Sys Study – Levels of Test and Int
- Combined Error Anal.
- Economic Benefits
- Experiments Usage
- Exp[illegible]t – Instrument Module Inter. Reqmts

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